ADVANCED PLANETARY PROBE

FINAL TECHNICAL REPORT

TO THE JET PROPULSION LABORATORY

VOLUME 1
STUDY APPROACH

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TRW SYSTEMS

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ERRATA

These pages give corrections to be inserted in "Advanced Planetary Probe Study, Final Technical Report," submitted under Jet Propulsion Laboratory Contract 951311 on July 27, 1966, by TRW Systems, One Space Park, Redondo Beach, California.

CORRECTIONS TO VOLUME 1

Location	Correction
p 24, Figure 2	In the third box from the top, replace "GENERATIONS..." by "GENERATION..."
p 34, Table 3	The entry under "Operational Turn ON/OFF" corresponding to "Radio Propagation" should read: "Turn on at launch. Use to limit of transmitted signal."
p 73, second paragraph	In the second sentence change "Venus" to "Uranus"

CORRECTIONS TO VOLUME 2

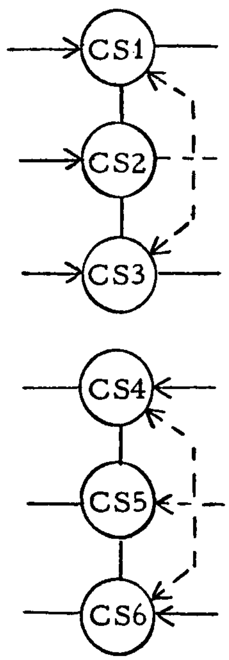
Location	Correction
p 22, Figure 2-9 caption, and p xi	Augment, so as to read, "Earth-Jupiter 1972 Trajectories, C_3, Geocentric Launch Energy"
pp 23 to 27, Figures 2-10 to 2-14	Change value of largest Type I C_3 contour from 130 to 140
p 49, before last paragraph	Insert heading, "2.2.8 Delineation of the Launch Period"
p 52, before first paragraph	Insert heading, "2.2.9 Characteristics of the Launch Period"
pp 52, 57, 60, 67, 68, and 70	Renumber the headings as follows: 2.2.8 becomes "2.2.10 Sample Trajectory" 2.2.9 becomes "2.2.11 Encounter Geometry" 2.2.9.1 becomes "2.2.11.1 Mission Objectives" 2.2.9.2 becomes "2.2.11.2 Spacecraft Design Constraints" 2.2.9.3 becomes "2.2.11.3 Sample Encounter Trajectories" 2.2.10 becomes "2.2.12 Trajectory Accuracy"
p 75, Figure 2-38	Note A: Replace "REFERENCE" by "(WHIPPLE)" Note B: Replace "REFERENCE" by "(DUBIN AND MC CRACKEN)" Note C: Replace "REFERENCE" by "(WHIPPLE)"

CORRECTIONS TO VOLUME 2 (CONTINUED)

Location	Correction
p 75, Figure 2-38	Curve E should be relocated 1/2 division above its present location, so as to be 1 division (or a factor of 100) above Curve C
p 76, Equations	The two equations should be labeled A and B, respectively
p 77, Equation	The equation should be labeled C
p 78, 3 lowest equations	The three lowest equations should be labeled D, E, and F, respectively
p 78, Second equation	This equation, D, should be written "log N = -1.34 log M - 12.18"
p 81, Figure 2-39	In left figure, extend arrow labeled "V_{mr}" to heavy line
p 123, Figure 4-11	Abscissa signs should be reversed: -4, -8, and -12 above the zero line, and +4, +8, +12 below the zero line
p 125, Figure 4-12	Dashed curve should be labeled, "B·T, B·R CONTROL"
p 129, Figure 4-14	Longest heavy line should be marked "0.1." Add note that numbers refer to days after launch.
p 226, Figure 4-52	Note should read "BASELINE DESIGN IS 7.1 TO 8 LBS HEAVIER THAN MINIMUM WEIGHT STRUCTURE"
p 228, second line	Replace parenthetical note by "(flux F of Table 4-8)"
p 312, Figure 7-17 p 315, Figures 7-18 and 7-19	Ordinate axis should be labeled "B·R (10^3 KM)" and values above the zero line should be negative: "-10," "-20," etc. Abscissa axis should be labeled "B·T (10^3 KM)" and values to the left of the zero line should be negative: "-10," "-20," etc.
p 319, Figure 7-20	In upper right-hand corner, add fraction line to make integral read $\int \left\| \frac{1}{r^2} \text{SIN } 2\,\zeta \right\| dt$ In the caption, replace "$1/r^2$ sin 2 ζ" by "$(1/r^2)$ sin 2 ζ"

CORRECTIONS TO VOLUME 2 (CONTINUED)

Location	Correction
p 319, Figure 7-20 (continued)	The left-hand scale should be identified $\frac{\text{SIN } 2\zeta}{r^2}\ (1/AU^2)$
	In the tabulation in the upper right-hand corner of values of the integral for the different areas, replace "DAYS" by "DAYS/AU^2"
p 321, Figure 7-21	Replace the symbol "ξ" by "ζ"
p 324, Figure 7-23	Reference should be changed from "(SEE FIGURE 3.6.7-4.)" to "(SEE FIGURE 2-39.)" Change "V_m" and "V_M" to "V_{mr}"
p 325	Change "V_m" to "V_{mr}" (two places)
p 411, Figure 8-26	Insert vertical lines between pairs of circulator switches: CS1 and CS2; CS2 and CS3; CS4 and CS5; CS5 and CS6. These interconnections should appear thus:



Location	Correction
p 418, Figure 8-28	Extend one arrow from note, "SPACECRAFT HIGH GAIN" so that the note applies to the three right-hand curves
p 426, Figure 8-30	Insert arrows from note, "SUM OF NEGATIVE TOLERANCES = 2.9 db" to two highest curves

CORRECTIONS TO VOLUME 2 (CONTINUED)

Location	Correction
p 532, Table 12-3	In the first column, under "Magnetometer" insert "Micrometeoroid"

CORRECTIONS TO VOLUME 3

Location	Correction
p 7, Figure 1, and p vii	In caption, replace "Relating" by "Related"
p 32, last paragraph	Replace "V_a" by "r_a"
p 32, first equation	Replace "δ" by "g"
p 90, first paragraph of 4.2	Replace first sentence by "In this section the functional requirements imposed on the design of the spacecraft for the 1972 earth-Jupiter flyby mission are examined and extended to cover the period of one Jovian year, from the 1968-1969 opportunity to the 1980-1981 opportunity."
p 91, second line from bottom	Replace "1969 to 1970" by "1969-1970"
pp 96-98	The entire section starting on page 96 with the second paragraph: "4.2.3 Encounter Geometry The variation of the characteristics..." and ending on page 98 with the first paragraph of 4.25: "...of these requirements in Volume 2 for the 1972 mission." should be removed from this location and inserted intact on page 112 after the first complete sentence and before the heading, "4.2.6 Requirements Imposed by Science Payload." Thus the material between pages 98 and 112 is all part of Section 4.2.2.
p 100, last paragraph, fourth line p 100, last line	Change "United States" to "versus"

CORRECTIONS TO VOLUME 3 (CONTINUED)

Location	Correction
p 112, Figure 38 caption, and p ix	Change "Versus" to "of"
p 130, Figure 52, and p ix	Caption should read "Earth-Neptune 1978 Trajectories, C_3, Geocentric Launch Energy." A label should be added to the upper right-hand corner to indicate that the numbers 10,350; 9,750; etc., are "FLIGHT TIME, DAYS."
p 174, Figure 94	Change "SATURN VIA JUPITER SWINGBY, 1979" to "SATURN VIA JUPITER SWINGBY, 1978"
p 200, third line	Second sentence of the paragraph should read "An error in the direction of $\overline{B}$ causes..."
p 202, Table 19	After "$\delta\omega$:" insert "(deg)." After "...by i:" insert "(deg)"
p 219, 17th line	Replace "of" by "at a" to give "...the sensible atmosphere at a grazing angle and..."

CORRECTIONS TO VOLUME 4

Location	Correction
p v, Figure A-6 caption	Change "$2\nu^2\alpha$, E" to "$2\nu^2\alpha\epsilon$"
p 6, Figure A-4 p 7, Figure A-5 p 202, line 9 p 209, second, third, and last equations p 210, first and fourth equations p 217, second and fourth equations, and line 15 p 13, Figure B-1	Replace "ξ" by "ζ"
p 13, Figure B-1	Interchange "x" and "y" in the figure. The equation for "b'(θ)" should be revised: "$b'(\theta) = \frac{\partial b(\theta)}{\partial \theta}$"
p 16, Figure B-3	Replace "y^2" by "y_2"
p 203, Figure K-1	Change symbol "n_{arr}" to "η_{arr}"
p 203, last line	Replace "V_p" by "V_p^2"

CORRECTIONS TO VOLUME 4 (CONTINUED)

Location	Correction
p 206, second equation	Add fractional line within parentheses to give: $$"\Delta B = \left(\Delta S_1 - \Delta S_2 \frac{V_j}{V_{arr}}\right) \frac{V_{arr}}{V_{rel}} \sin \theta_j"$$
p 208, line 3	Replace "$\sin 2\xi / r^2$" by $\left(1/r^2\right) \sin 2\zeta$"
p 212, last line	Change "Δ_i" to "Δi"
p 213, first line	Change "$Vi \cos \theta_i$" to "$V_i \cos \theta_i$"

CORRECTIONS TO VOLUME 5

Note: Although Volume 5 is classified (CRD) this errata sheet is unclassified, unless it is attached to Volume 5.

Location	Correction
p iii	Change Item 3 to "3. APPLICABILITY OF SNAP-27 AND SNAP-19"
p 1, line 5	"Snap-27 and Snap-29" should be changed to "Snap-27 and Snap-19"
p 6, Table J-3	In the last line, change "raw power bias" to "raw power basis"
p 7, Figure J-1	In the top view, change the dimension "3.0 FT" to "3.0 IN."
p 16	The last element of the first line of the decay chain at the bottom of the page should be changed from $\xrightarrow[51.5\ \text{sec}]{}$ to $\xrightarrow[51.5\ \text{sec}]{\alpha}$
p 21, Table J-8	Note a should end "Reference 21." instead of "Reference 15"
p 21 and p iii	The heading of Section 5.2 should be "Snap-19 Radiation Fields"
p 23 and p iv	Title of Figure J-7 should be "Neutron and Gamma Dose Rates from the Snap-19 Radioisotope Thermoelectric Generator"

CORRECTIONS TO VOLUME 5 (CONTINUED)

Location	Correction
p 26 and p iv	Title of Figure J-11 should be "Neutron and Gamma Dose Rates from the Snap-27 Radioisotope Thermoelectric Generator"
p 26, Figure J-11	Two curves are labeled "NEUTRON DOSE RATE (3.33×10^4..." The lower of these curves should be relabeled "NEUTRON DOSE RATE (4.2×10^3..."
p 30	Reference 17 should be changed to "Strominger, D., et al, ..." Reference 20 should be changed to "Evans, R.D., ..."

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ADVANCED PLANETARY PROBE STUDY

FINAL TECHNICAL REPORT

27 July 1966

Prepared for the Jet Propulsion Laboratory
under Contract 951311

Volume 1

Study Approach

This work was performed for the Jet Propulsion Laboratory, California Institute of Technology, sponsored by the National Aeronautics and Space Administration under Contract NAS7-100.

TRW SYSTEMS
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ILLUSTRATIONS

TABLES

1. INTRODUCTION

The results of TRW's study of an Advanced Planetary Probe for the Jet Propulsion Laboratory are described in five separately bound volumes:

Volume 1. Study Approach

Volume 2. Spin-Stabilized Spacecraft for the Basic Mission

Volume 3. Alternate Spacecraft and Missions

Volume 4. Appendixes

Appendix J. RTG Considerations (Confidential)

Here in Volume 1 the work performed in establishing the methods of approach to the study is described, including a detailed discussion of the scientific objectives of conceivable missions. The concept of the basic mission, Jupiter flyby with a 50 pound science payload, is established and the other types of possible missions are evaluated.

Since primary emphasis in this study is on a Jupiter flyby mission, Volume 2 and a large portion of Volume 3 describe spacecraft systems suitable for such a mission. Volume 2, the largest portion of the report, is a detailed description of a 500-pound, spin-stabilized spacecraft which can carry 50 pounds of scientific payload on a flyby of Jupiter using the Atlas/Centaur booster with a solid propellant third stage, TE-364-3. The spacecraft can transmit 700 bits/sec from the vicinity of Jupiter. It has no basic lifetime limitations, which means that it could operate long after the flyby, and the spacecraft should swing well beyond 10 AU. The estimated reliability of the spacecraft is 0.79 for two years, but experience with other spacecraft indicates that once the spacecraft has operated a few days in a space environment it will continue almost indefinitely. Volume 2 provides the basis for the succeeding sections, in that only the differences between this spacecraft concept and alternative concepts are discussed in the later sections.

Section 2 in Volume 3 describes a 575 pound, 3-axis stabilized spacecraft which can also perform the Jupiter flyby mission. Although the reliability is somewhat lower than that of the spin-stabilized probe, the stabilized spacecraft provides an important scientific advantage in that the TV experiment can achieve better resolution than can be achieved from a spinning spacecraft. This spacecraft can also be launched using the Atlas/Centaur/TE-364-3 combination.

Section 3 in Volume 3 discusses spacecraft with three alternate science payload weights; 12, 100, and 250 pounds. The spacecraft with the 12 pound payload weighs about 275 pounds. Since there is no low cost booster which could launch this vehicle efficiently, it could be used with the Atlas/Centaur/TE-364 combination for a shorter flight time mission. Spacecraft configurations for the 100 and 250 pound payloads have been examined in sufficient detail to provide a comparison of overall spacecraft weight, scientific objectives, booster requirements, etc., with the 50 pound payload concepts.

To ensure that no critical trajectory characteristics were overlooked, the spacecraft concepts were studied for a specific launch year, 1972. Section 4 in Volume 3 discusses the differences for missions during the years 1973 to 1980 and identifies the effects the changed requirements will have upon the spacecraft. Section 5 discusses flyby missions to planets beyond Jupiter. This section uses both the spin-stabilized and 3-axis controlled 50 pound science payload spacecraft as bases. The effects of the increased lifetime, thermal changes, and other factors are described. Section 6 discusses the growth capability of the 50 pound science payload spacecraft to orbiter missions. Trajectories, encounter geometry, deboost requirements, and in-orbit requirements are analyzed, and the adaptability of the 50 pound payload spacecraft to the concept is evaluated. Section 7 is a very brief description of the problems of a capsule entry mission to the planet study. The difficulties of this mission have made this analysis of secondary importance.

The cost effectiveness of the 500 pound, spin-stabilized concept is evaluated at the end of Volume 2. Volume 3 concludes with a general discussion of cost effectiveness applicable to all of the concepts covered in the study.

Additional details on various topics, particularly with respect to Volume 2, are included in the appendixes of Volume 4. Because of its classification, the detailed discussion of RTG's as applicable to the Advanced Planetary Probe is bound separately, as Appendix J.

2. DEFINITION OF TASK

2.1 STATEMENT OF WORK

The purpose of the Advanced Planetary Probe study is to perform a conceptual design and feasibility study to develop first-generation spacecraft concepts adaptable for long range, long duration planetary missions in the region beyond Mars. The work statement, which is reprinted in part as Appendix A to this volume, defines the missions of direct applicability of the spacecraft system designs to be basic flyby missions of the planets Jupiter, Saturn, and Neptune. In addition, the growth potential of these conceptual designs to extend to orbiter and planetary capsule entry missions is to be examined. The scientific objectives of these missions are the following:

- Measurement of the spatial distribution of interplanetary and planetary particles and fields
- Measurement of the salient features of planetary atmospheres, with particular emphasis upon remote measurements from a flyby spacecraft.
- Observations of the planets, i.e., visual, infrared, etc.

In addition to this description of the missions and their objectives, the work statement also implies the method by which the conceptual design is to be synthesized: by recognition of a hierarchy of requirements stemming from the mission objectives and extending down to the subsystem level. The depth to which the design concepts are to be detailed, analyzed, and verified for feasibility is not explicitly stated; however, the format for reporting and describing the systems and subsystems indicates that particular attention is to be devoted to those areas of technical feasibility which are unique to the nature of Advanced Planetary Probe missions and to the technical approaches by which the mission requirements are met.

2.2 PRIORITY LISTING OF RESULTS

The organization of the report, as well as the study itself, reflects a priority listing by JPL personnel delineating the relative order of importance of the sections of the study. It also coincides with an orderly method of studying a broad range of missions and of the parameters that describe missions, by concentrating at first on a single approach to a single mission, and using this as a point of departure from which to address the other aspects identified in the work statement.

2.2.1 Spin-Stabilized Spacecraft for the Basic Mission

Volume 2 is devoted entirely to the conceptual design and feasibility study of a spin-stabilized spacecraft for the basic mission, a 1972 Jupiter flyby mission with a 50 pound science payload. A major portion of the entire study and the final report are devoted to this subject for two reasons. We have proposed and adopted spacecraft design constraints for this study— radioisotope power, extremely high-gain spacecraft antenna, and earth-oriented cruise attitude — as discussed in Section 3. For a spacecraft whose mission is the exploration of interplanetary and planetary environments at great distances from the earth, these concepts and spin stabilization are sufficiently novel, in comparison with current space technology, that a number of the contributing facets of this concept must be addressed and studied to describe adequately what the concept consists of, and to verify to the extent possible by analytical methods, the feasibility of these facets and of the integrated design. Thus, for the basic mission, the spacecraft approach outlined in this study, with spin-stabilization, is emphasized to a degree not necessary for a more conventional concept which would be adequate for a less challenging mission.

The second reason for the emphasis on this mission is that, having established a firm basis for satisfying the objectives of a Jupiter flyby in a particular year, with a 50 pound science payload, the parametric mission variations which are to be studied may be done so by comparison with this basic case. Therefore, although the other missions identified by the work statement imply a complexity comparable to the basic mission and a breadth encompassing more dimensions, the amount of the report devoted to these other missions is reduced in comparison, because each subsequent section of the report (Volume 3) treats its subject by amendment and increment to the preceding sections. These extensions from the basic mission are identified in the following paragraphs.

2.2.2 Three-Axis Stabilized Spacecraft for the Basic Mission

Second in priority and in sequence is the conceptual design of a 3-axis stabilized spacecraft addressing the same Jupiter flyby mission considered for the spin-stabilized spacecraft. This section serves two purposes, both of which are provided by comparisons. The first comparison, between this spacecraft design and the design of current spacecraft to planets close to the earth, e.g., Mariner 4 to Mars, indicates the consequences of simultaneously imposing the characteristics of missions to the outer planets and the spacecraft design constraints adopted for this study which are independent of the mode of attitude control.

Comparison is also afforded between this spacecraft and the spin-stabilized spacecraft for the same mission, pointing out the implications and consequences of the two modes of attitude control. The results of this second comparison will also influence the extension to mission variations addressed in this study.

2.2.3 Variations in Science Payload

The 50-pound science payload adopted in the basic mission is by no means the only complement of scientific instruments to be considered for Jupiter flyby missions. Not only is the allocation of a 50-pound payload to particular instruments open to review and alteration, but the appropriate total weight to be devoted to scientific instruments is by no means fixed. In fact, it is hoped that the results of this study may serve to indicate what ranges of payload weights are most efficient and effective in achieving the mission objectives.

However, recognizing that this study is addressed to first-generation spacecraft concepts, the maximum payload considered has been arbitrarily set at 250 pounds, certainly not an unduly restrictive limit. At the other end of the spectrum, a minimum science payload of 12 pounds was included, representing what is deemed to have a threshold scientific value.

These variations in science payload are considered in the context of a 1972 Jupiter flyby mission. Both the spin and 3-axis stabilization concepts for attitude control are considered for applicability.

2.2.4 Other Launch Opportunities

As the work statement fixes the time period for Advanced Planetary Probe mission accomplishments to be 1970 to 1980, it is necessary to examine the variation in mission characteristics and requirements which accrue from an alteration of the launch year. It is noted that this 11-year period is only one year short of a complete revolution of Jupiter about the sun. Therefore, the trajectory characteristics are examined for a 12-year period, and provide essentially a complete cycle of earth-Jupiter opportunities which may be expected to repeat in later decades. Specifically, those aspects of trajectory geometry which are significant to the 1972 Jupiter mission are examined for launches throughout the decade. The spacecraft designs conceived in the preceding sections are then interpreted for applicability for all launch opportunities.

2.2.5 Planets Beyond Jupiter

The work statement specifically identifies Saturn and Neptune as additional target planets. By implication, the planet Uranus is logically included as being similar to the other large outer planets, and intermediate in location. Many of the concepts presented will also be applicable to missions to Pluto, but the dissimilarity of this planet and its orbit to the others reduces the value of such an extension.

2.2.6 Orbiter and Capsule Entry Missions

The final portion of the work statement and organization of the report is the study of growth capabilities to orbiter and capsule entry missions at Jupiter and at the farther planets. More attention is directed to the extension of the spacecraft designs to the performance of orbiter missions than to their adaptations for the delivery of capsules. From the starting point of a flyby spacecraft, it is a more direct extension to outline the objectives of an orbiter mission than it is in the case of a capsule mission. Basically, the planet orbiter remains in the same environment and makes the same classes of observations that the flyby spacecraft does; the principal change is the extended duration in the proximity of the planet. However, for the capsule entry missions, the nature of the measurements and, indeed, the scientific objectives themselves may be directed towards substantially different goals.

In addition, the ultimate feasibility of the extension of the design concepts to orbiters, while involving more stringent requirements on some of the subsystems, is unquestioned; however, the technological basis for achieving meaningful capsule entry missions, particularly in the instance of the massive planets, Jupiter and Saturn, is anything but clear. While we can define and address the growth of Advanced Planetary Probe missions to orbiters, it is beyond the scope of this study to do all the groundwork necessary to outline feasible capsule entry missions to which meaningful objectives and requirements can be attached.

3. DEFINITION OF TERMS

3.1 MISSIONS

In general, the term "mission" is used in this report to encompass and describe the events which are associated with directing one or more spacecraft from the earth and which terminate with the accomplishment of the objectives. Although a mission may be general, pertaining to several target planets or to several launch years, it is usually associated specifically with a single planet and a single launch opportunity. (A "program" consists of a series of missions with a progression of related objectives and employing consanguine spacecraft design concepts.)

The basic differences between missions have to do with the objectives of the missions and the major means by which these objectives are achieved. These major classes are defined:

Basic Mission	In the basic mission of this study, the spacecraft is launched from the earth during the 1972 earth-Jupiter opportunity, and is directed on a trajectory which takes it close to the planet Jupiter. In addition, the science payload for the basic mission is fixed at 50 pounds.
Flyby Mission	In a flyby mission, the spacecraft passes close to the target planet. No propulsion forces are employed to alter the trajectory so as to remain in the vicinity of the planet, and the spacecraft departs from the region of the target planet, although its trajectory will have been perturbed.
Orbiter Mission	In an orbiter mission, approximately at the time when the spacecraft is closest to the target planet, its trajectory is deliberately altered by on-board propulsion so that it remains in an orbit about the target planet as a satellite.

Capsule Entry Mission	In a capsule entry mission, a capsule is detached from the basic spacecraft, and directed to impinge on the atmosphere of the planet. There is not necessarily any intention that the capsule descent intact to the surface of the planet. The spacecraft, after the capsule is separated, may itself pursue either a flyby or orbiter mission.

3.2 SYSTEMS

For purposes of organization of the elements of a mission, or a program of several missions, we identify the following systems as components of the whole endeavor:

Launch Vehicle	The launch vehicle includes the multistage boost vehicle which injects the spacecraft onto an interplanetary trajectory (but excluding a solid injection stage, if used), including the nose fairing which protects the spacecraft, and all hardware up to the field joint where the spacecraft is mated. Generically, the launch vehicle system also includes all appropriate ground support and test equipment.
Spacecraft System	The spacecraft system encompasses the spacecraft itself and all component subsystems, the science payload, the adapter which is mounted to the launch vehicle, and the solid injection stage, if one is used. The spacecraft system generically includes appropriate ground support equipment, operational support equipment, operational support equipment, test equipment, etc.
Launch Operations System	The launch operations system does not include any flight hardware, but constitutes the operational responsibility for supporting and conducting the launch of the combined launch vehicle and spacecraft through the separation of the spacecraft from the launch vehicle.
Mission Operations System	Operational responsibility for supporting and conducting the mission after the spacecraft is separated from the launch vehicle is borne by the mission operations system.

3.3 MISSION EVENTS

In the analysis of the various missions throughout the study the following terms are used:

Prelaunch	Collectively, all events before liftoff
Launch	Collectively, all events from liftoff to injection
Liftoff	Departure of the combined launch vehicle-spacecraft from the ground
Injection	Final termination of thrust of the last stage of the launch vehicle (or of the solid injection stage, if used)
Separation (Shroud)	Detachment of the nose fairing from the launch vehicle during ascent.
Separation (Spacecraft)	Detachment of the spacecraft from the spacecraft-launch vehicle adapter after injection. (Where the spacecraft carries a solid injection stage, spacecraft separation occurs before injection.)
Orientation Maneuver	A programmed alteration of the spacecraft attitude to cause it to assume a desired orientation
Reorientation Manuever	A programmed alteration of the spacecraft attitude to cause it to return to the cruise orientation.
Midcourse Trajectory Correction Maneuver	A propulsive maneuver performed to compensate for inaccuracies or perturbations so as to redirect the spacecraft toward the intended aiming point. Generally, it requires orientation to a specific attitude, operation of the rocket engine, and reorientation to the cruise attitude. The time of this maneuver is during the interplanetary flight, but not necessarily at the midpoint.
Encounter	Generally, encounter encompasses events occurring when the spacecraft is near the target planet. Specifically, it refers to the time when the spacecraft is at its point of closest approach (periapsis).

Orbit Insertion	The propulsive braking maneuver by which the (orbiter) spacecraft trajectory at the target planet is changed from approach (hyperbolic) to orbital (elliptical).

3.4 TRAJECTORY TERMS

In dicussing the trajectories possible for the various missions studied, the following terms are used:

Direct Trajectory	An interplanetary trajectory from the earth to a target planet, in which no intermediate planets (or satellites) are approached closely enough to significantly influence the trajectory.
Swingby Trajectory	An interplanetary trajectory from the earth to a target planet, in which an intermediate planet is passed sufficiently closely to exploit the effect of its gravitational attraction. This exploitation may provide reduced mission duration, reduced launch energy, or opportunity for intermediate scientific observations.
Launch Opportunity	The time during which trajectories to a target planet may be initiated from the earth, with reasonable launch energies. A launch opportunity is usually identified by the year in which it occurs, and the target planet.
Launch Period	The space in arrival date-launch date coordinates in which earth-planet trajectories are possible, in a given launch opportunity; specifically, the number of days from the earliest possible launch date to the latest.
Launch Window	The time in hours during which a launch is possible on a particular day.
Geocentric (heliocentric; planetocentric)	Described or measured with respect to inertial coordinates centered with the earth (sun; planet). Pertaining to the portion of the flight in which the trajectory is dominated by the gravitation of the earth (sun; planet).
C_3, Launch Energy, Injection Energy	Twice the geocentric energy per unit mass, of the injected spacecraft. This is equivalent to the square of the geocentric asymptotic departure velocity.

Asymptote	The linear portion of a hyperbolic (escape) trajectory which is approached and approximated at large distances from the attracting center
DLA	Declination of the outgoing geocentric launch asymptote
ZAL	Angle between the outgoing geocentric asymptote and the sun-earth vector
ZAP	Angle between the incoming planetocentric asymptote (at the target planet) and the planet-sun vector
ZAE	Angle between the incoming planetocentric asymptote (at the target planet) and the planet-earth vector
V_{HP}	Planetocentric asymptotic approach velocity
Parking Orbit	An unpowered, geocentric, approximately circular orbit, separating the twin powered portions of the launch and injection seqence.

4. ASSUMPTIONS AND CONSTRAINTS

4.1 DOMINATING MISSION CHARACTERISTICS

Before enumerating the assumptions and constraints adopted for this study, it is worth observing the characteristics which dominate Advanced Planetary Probe missions. In particular, the characteristics are reviewed which are substantially altered in comparison with analogous effects in contemporary spacecraft missions directed to the moon, Venus, and Mars, and which exert an important influence on spacecraft design and performance.

4.1.1 Extreme Distances from the Sun

To date, all spacecraft operations have been between 0.7 and 1.5 astronomical units (AU) from the sun. The Advanced Planetary Probe of the 1970's will have to travel to a distance of 5 AU from the sun to reach Jupiter, 9 AU to Saturn, and 30 AU to Neptune. At these distances from the sun, three major effects are noted:

a) The radiant energy from the sun which may be intercepted by the spacecraft and converted into electrical power decreased as the square of the distance from the sun. Thus, in comparison with operation at the earth, the watts per square foot available is down a factor of 25 at Jupiter, and a factor of 900 at Neptune.

b) The radiant energy of the sun as a factor in thermal control undergoes the same square law decrease. Not only must the spacecraft be capable of maintaining adequate component temperatures when the solar input has dropped to an almost negligible quantity, it must also accommodate the variation in intercepted solar energy from 1 AU out to the distances. Thus, the design must conserve heat when far from the sun, but also reject heat when close to the sun.

c) The earth-spacecraft-sun angle decreases as the spacecraft reaches more remote regions of the solar system. The maximum elongation of the earth's orbit is 12 degrees, when seen from Jupiter, 6 degrees at Saturn, and 2 degrees at Neptune. Of course, from any point in the outer solar system, the earth's annual motion takes it back and forth between eastern and western elongations. But the restriction on maximum angles between the sun and the earth have an influence on spacecraft attitude and communication considerations.

4.1.2 Extreme Distances from the Earth

As the spacecraft penetrates the outer regions of the solar system, there is a concomitant increase in the distance from the earth. From Jupiter the distance is 4 to 6 AU, and for Saturn and Neptune, the distance is also within 1 AU of the sun-planet distance. The dominating effect here is on the performance of the communications system. To achieve the same data transmission rates which other spacecraft have demonstrated for ranges of the order of 1 AU requires some combination of increases in spacecraft transmitter power, spacecraft antenna gain, and ground receiver antenna gain equivalent to the square of the distance in AU.

A second consequence of the large earth-spacecraft distances is the appreciable time lapse for round trip communications. Even at the smallest earth-Jupiter separation it takes 1.1 hours for a command to be transmitted to the spacecraft and a response to be received at the earth. At the distance of Neptune this round trip time delay exceeds 8 hours. Obviously, conventional concepts of command and control must be carefully reviewed before they are applied in these circumstances.

4.1.3 Mission Duration

The transit times for the spacecraft to travel from the earth to the target planets are very long compared to planetary missions conducted to date. Minimum-energy or Hohmann transfers range from two to three years for Jupiter missions, and up to 30 years to Neptune. These times are reduced by employing more than the minimum injection energy at launch from the earth, or, in the case of missions to the most distant planets, using swingby trajectories about Jupiter or another intermediate planet. Nevertheless, it will inevitably require two years or more to reach Saturn and six years or more to reach Neptune, even with large increases in launch vehicle capability above the minimum requirements.

The long durations of the missions have three principal consequences, which should influence program planning as well as spacecraft design:

- Design simplicity of the spacecraft and reliability of the operation of its subsystems are of paramount importance in achieving reasonable probabilities of success.

- The program must recognize a long time cycle within which the results of one planetary mission can influence the design and objectives of a later mission. This may lead to a leap-frogging of mission scheduling such that several missions in a sequence must be formulated open loop, that is, without the benefit of early results.
- The operational cost of supporting the spacecraft mission during transit time is increased.

4.2 SPACECRAFT CONSTRAINTS

For the Advanced Planetary Probe study, TRW has proposed three general spacecraft design constraints to accommodate the mission characteristics discussed above. These constraints, which embody an approach to satisfy the challenging requirements of missions to the outer planets of the solar system, have been adopted as ground rules within which the study is conducted, and for which internal justification—by means of direct comparisons—is not offered.

4.2.1 RTG Power

Radioisotope thermoelectric generators (RTG's) have been selected for spacecraft power for the Advanced Planetary Probe because of inadequate, or at least extremely inefficient, solar power available at the large distances from the sun. In addition, the other spacecraft constraints mitigate against a dependence on the solar orientation of the spacecraft. Among other nonsolar forms of power which might be considered, nuclear reactor power is rejected as it is suitable only for substantially larger power systems, and the concept of radioisotope thermionic electricity generation is not sufficiently mature at this point.

The adoption of RTG power is accompanied by penalties which are recognized. There is a strong interaction with the other subsystems of the spacecraft, due to RTG heat rejection requirements and the nuclear radiation environment imposed on experiment sensors. In addition, the cost of radioisotope fuel is a significant factor, and the necessity of satisfying aerospace safety requirements must be considered.

4.2.2 High-Gain Spacecraft Antenna

To meet the challenge of a satisfactory communication link at extreme spacecraft-earth distances a high-gain spacecraft antenna consisting of a large body-fixed paraboloidal reflector is proposed for primary downlink communications. Although several means of achieving high gain in the spacecraft antenna have been investigated as candidates, the selected approach makes use of deployable, lightweight honeycomb sandwich panels, permitting the formation of a rigid dish with a diameter greater than that of the nose fairing, and weighing approximately 0.25 lb/sq ft.

Where communications data transmission rate requirements are more modest, it is possible to use a reflector smaller than the nose fairing, and thus avoid the necessity of deploying panels.

A consequence of the use of the high-gain antenna is that accurate pointing attitudes must be maintained.

4.2.3 Earth-Oriented Cruise Attitude

The third constraint is that the spacecraft during the cruise phase of the mission (essentially all of the mission, except for midcourse correction maneuvers) shall be oriented so as to direct the high-gain antenna axis toward the earth. This third constraint is a necessary adjunct to the high-gain antenna for the achievement of high data transmission rates. It also complements the theme that the Advanced Planetary Probe, exploring portions of the solar system very remote from the sun, is essentially independent of the sun: solar radiation is not required for power generation, spacecraft thermal control mechanization must be relatively insensitive to solar radiation, and the attitude control of the spacecraft in cruise is based on the earth rather than on the sun.

4.3 BALLISTIC TRAJECTORIES

It is assumed for this study that primary propulsion to direct the spacecraft to the target planets is accomplished by the launch vehicle (augmented by the solid injection stage, if used). The spacecraft propulsion system is employed only for the removal of small errors in the interplanetary trajectory, and for orbit insertion, in the case of an orbiter mission. By this assumption, the study does not consider missions in which low thrust spacecraft propulsion, e.g., electric ion propulsion, is used to carry the spacecraft outward from the sun.

By restricting the study to ballistic trajectories, a possible trajectory advantage is surrendered in that low thrust trajectories may be accommodated over a somewhat longer launch period. On the other hand, the consideration of low thrust trajectories imposes electric power requirements which are completely inconsistent with the use of radioisotopic power, and, indeed, strain the capabilities of either nuclear reactor power or solar photovoltaic power. In addition, the state of development of electric ion propulsion systems is so immature that it invites doubt as to its applicability during the decade of the 1970's.

4.4 TIME PERIOD FOR THE MISSIONS

The decade 1970 to 1980 is prescribed by the work statement as the period of mission accomplishments. For purposes of subsystem design, a reasonable lead time is necessary so that the design is established sufficiently in advance of the launch date. For this purpose it is felt that the earliest launches in the decade must be accomplished with hardware which essentially reflects the 1966 state of the art. The later launches may reasonably be based on developments in technology expected to occur at appropriately later times.

If the decade specified is interpreted to apply to the launches of spacecraft, the end of the mission, if determined by arrival at the more distant planets, could be as much as eight years later. It is noted in particular that launches for Jupiter swingby trajectories to all the outer planets are opportune in 1978 and 1979, leading to arrivals at Neptune as late as the middle of the 1980's.

4.5 AVAILABLE LAUNCH VEHICLES

Six launch vehicle combinations have been identified by memorandum from the Jet Propulsion Laboratory as candidates for the Advanced Planetary Probe study. The smallest of these, the Atlas/Centaur, has inadequate performance for any mission to Jupiter, and therefore is not adaptable to the Advanced Planetary Probe requirements. The largest is the

Saturn V/Centaur, with a capability grossly exceeding the reasonable and efficient requirements of even a large first-generation spacecraft. Because of this wide spread of the characteristics of the launch vehicle combinations suggested, we have conceived two additional combinations which provide intermediate performance capabilities in the ranges which can be exploited by Advanced Planetary Probe missions. These two launch vehicles, obtained by incorporating a solid rocket injection stage within the spacecraft system design, are the Atlas/Centaur/TE-364 and the Titan III/Centaur/TE-364. All of the launch vehicles are identified, and their performance capabilities are described in Section 2.1 of Volume 2.

The approach to this study implied by the work statement is that the spacecraft system design is based on requirements arising with the mission objectives and applied by extension to the spacecraft system and to its subsystems. It is further implied that the selection of the launch vehicle is subsidiary to the design of the spacecraft. On the other hand, it is certainly desirable to recognize that a limited number of identified launch vehicles provides a step function capability, and not the continuous spectrum of performance which would be available if one could specify a launch vehicle system to be designed to order for the mission. With these discrete levels of performance capability, as indicated in Figure 1 for a typical mission, it is seen that certain values of the spacecraft weight give a more favorable utilization of the launch vehicle capability than others. For example, spacecraft weights indicated by the letters A, B, and C, can all be accommodated for this mission by the same launch vehicle combination. It is evident that the minimum weight, A, makes inefficient use of this launch vehicle, since an only slightly lower spacecraft weight would be served by a smaller, less expensive booster. Spacecraft with weights B and C make most efficient use of the launch vehicle; while C is slightly greater than B it is probably less desirable, since it gives no margin for growth or for flexibility in accommodating changes in spacecraft payload.

Since cost effectiveness is one of the criteria to be examined in arriving at the optimum spacecraft design concepts, and since the efficient utilization of launch vehicle performance capability depends on

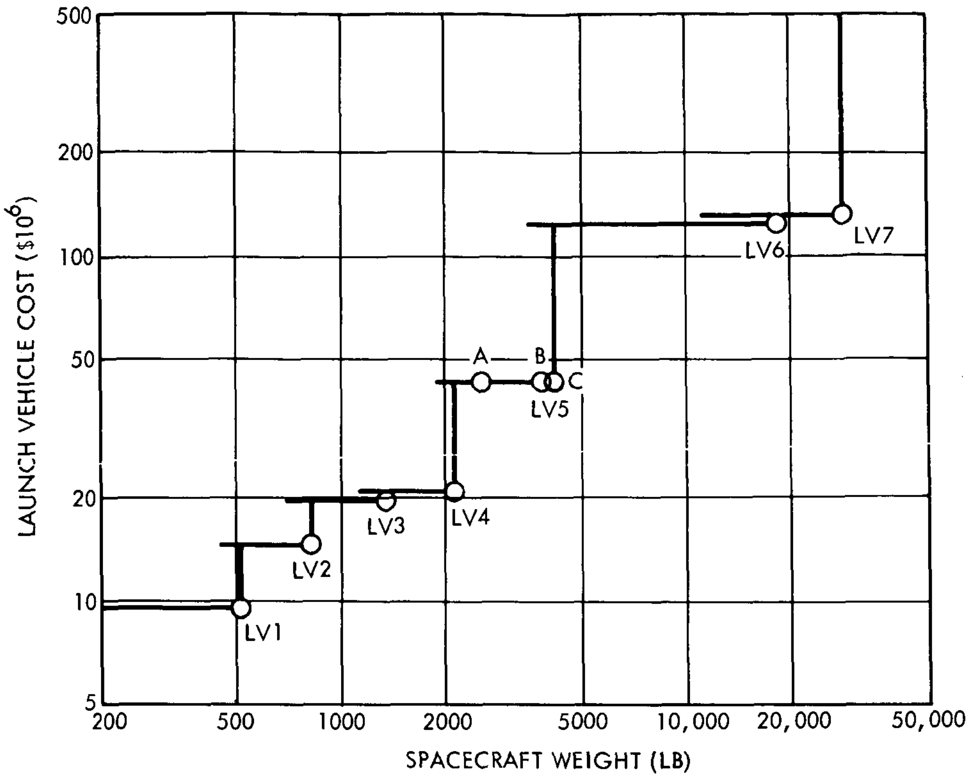

Figure 1. Launch Vehicle Cost versus Spacecraft Weight (C_3 = 100 km^2/sec^2)

matching the spacecraft to the launch vehicle, a satisfactory spacecraft design must take into account the capabilities of the available launch vehicles. However, this taking into account is not necessarily done by applying launch vehicle capabilities as an a priori constraint. In fact, the entire system conceptual design is an iterative process, as discussed in Section 5. It is possible and appropriate to introduce the launch vehicle capabilities only at the end of each iteration, in the process of evaluating performance and effectiveness of the spacecraft design, and effecting the appropriate influences by the alterations imposed on the subsequent iteration. Therefore, in a sense, the launch vehicle constraint is applied a posteriori in influencing the efficiency of the spacecraft design.

4.6 LAUNCH SITES

In this study the only launch site considered is Cape Kennedy, Florida, of the Eastern Test Range. Other launch restrictions, principally based on safety and communications requirements, are discussed at appropriate points in the report.

4.7 COMPATIBILITY WITH DSN

A ground rule for the Advanced Planetary Probe study is that the spacecraft systems and their operations shall be compatible with the operational capabilities and characteristics of the Deep Space Network as it now exists, or is expected to exist in the appropriate time period, as described by JPL Technical Memorandum 33-83 and by other sources.

4.8 CONSTANTS EMPLOYED

In this study, the following constants pertaining to characteristics of the solar system have been employed:

Astronomical unit	1 AU = 149,599,000 km
Speed of light	C = 299,792.5 km/sec
Solar gravitation constant	$\mu = 4\pi^2 (AU)^3/yr^2$

Planetary radii (equatorial)

Jupiter	R_j = 71,400 km
Saturn	R_s = 60,400 km
Uranus	R_u = 23,500 km
Neptune	R_n = 22,300 km

Planetary gravitation constants, μ

Jupiter	$\mu = 126.71 \times 10^6 \ km^3/sec^2$
Saturn	$\mu = 37.92 \times 10^6 \ km^3/sec^2$
Uranus	$\mu = 5.788 \times 10^6 \ km^3/sec^2$
Neptune	$\mu = 6.8 \times 10^6 \ km^3/sec^2$

Planetary axial rotation periods

Jupiter	9h 53m
Saturn	10h 26m
Uranus	10h 42m
Neptune	15h 48m

While accurate ephemeris tapes were employed for analytical and integrated trajectories, the following approximate ephemeris data are useful for purposes of visualization.

Quantity	Planet			
	Jupiter	Saturn	Uranus	Neptune
a, semimajor axis, AU	5.2028	9.539	19.18	30.06
Perihelion distance, AU	4.9520	9.008	18.28	29.80
Aphelion distance, AU	5.4536	10.070	20.09	30.32
e, orbital eccentricity	.0482	.05162	.04431	.00734
i, inclination to ecliptic, deg	1.306	2.487	0.772	1.773
Ω, longitude of ascending node, deg	100.18	113.3	73.7	131.4
$\tilde{\omega}$, longitude of perihelion, deg	13.35	89.6	172.5	25.4

5. METHOD OF APPROACH

The method of approach to the generation of spacecraft design concepts for Advanced Planetary Probe missions is outlined here. The functional requirements of a mission are determined and the influence of these requirements on the generation of spacecraft design concepts is indicated. The influence of the broad range of mission variations on the spacecraft design concepts is also indicated.

The following mission variations are treated in this study:

- Variation in the year of the launch opportunity, over the years 1970 to 1980
- Variation in the target planet: Jupiter, Saturn, Neptune
- Variation in the class of mission: flyby, orbiter, capsule entry
- Variation in the science payload weight.

In addition, the spacecraft designs considered and described may be divided into two classes, depending on the basic mode of attitude control: spin and 3-axis stabilization.

In Section 2.2 it is observed that the organization of the study and of the report lead to a concentration on the first priority mission, the generation of a spin-stabilized spacecraft design for a Jupiter flyby in 1972, with a 50-pound science payload. The other mission variations (and the 3-axis stabilized approach) are treated successively by examining the alterations and variations from the basic case.

Although the basic scientific objectives of Advanced Planetary Probe missions are not altered, a parameter whose variations are studied is the weight devoted to the science payload. The range considered covers a minimum payload of 12 pounds, having minimum scientific value for an interplanetary and planetary flyby mission, to a maximum arbitrary weight at 250 pounds, which would establish such a great capability of planetary observations that it is more properly matched to an orbiter mission, perhaps, than to a flyby. Implicit in the variations of payload weight are variations in:

- To what degree of refinement and detail it is indended to satisfy the scientific objectives of the mission
- The nature of the mission requirements imposed by the experiments on the spacecraft system design
- The values of the resulting spacecraft design parameters—power, data, and scanning abilities—which make an optimum match with the experiment complement.

5.1 SPIN-STABILIZED SPACECRAFT FOR THE BASIC MISSION

The method of approach used in the generation and formulation of the spin-stabilized spacecraft system design for the 1972 Jupiter flyby mission, with a 50-pound science payload, the results of which are reported in Volume 2, is outlined in this section. A diagram summarizing the method is shown in Figure 2.

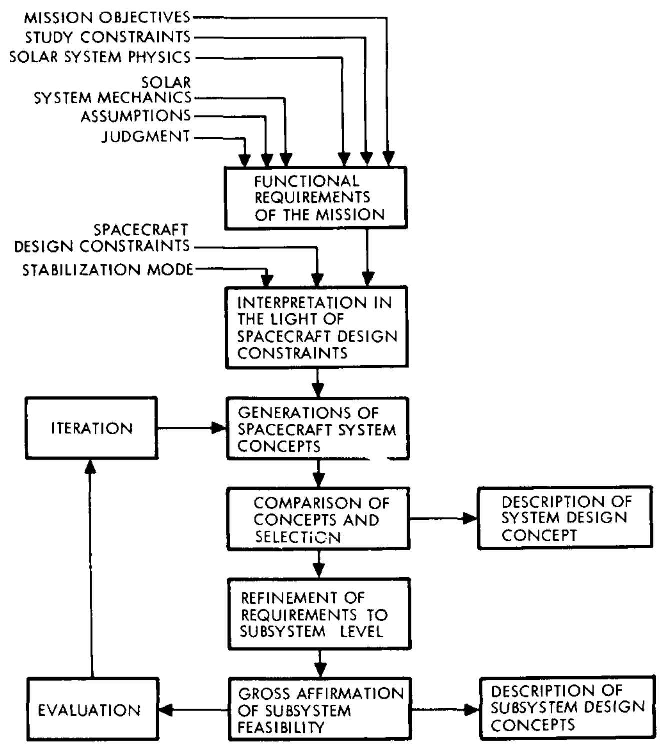

Figure 2. Iterative Approach to Spacecraft Design

It is important to observe that the method of approach employed, and described below, is an iterative one in which each iteration serves to refine the results of the preceding one, and bring them closer to the final results. However, Volume 2 does not report all of the iterations.

It does indicate the major alternates considered and the factors leading to the selections made, but primarily it is descriptive of only the final iteration of the process.

Because only the final iteration is described in detail, the evaluations of the proposed concept may, in some instances, appear "fixed", as though the design approach or the performance evaluation results were pre-conceived. However, these influences are frequently logically introduced by an interim evaluation performed at the conclusion of one of the intermediate iterations, and by this feedback process it affects subsequent iterations in a way which optimizes subsequent evaluations. An example of this influence, pertaining to the matching of the spacecraft system to available launch vehicles, was discussed in Section 3. 5.

5. 1. 1 Functional Requirements

The functional requirements which must be met by the spacecraft system design in accomplishing the basic mission are derived from a description of the mission itself, ground rules and constraints of the study, the physical and mechanical properties of the solar system, and by the introduction of other assumptions and judgment factors. These functional requirements may be classified according to the following list:

Launch Vehicles	The spacecraft design must be compatible with the performance characteristics of launch vehicles, with the dynamic envelope available, and with the mechanical and vibrational launch environment. The spacecraft must also be able to compensate for injection errors produced by the launch vehicle.
Interplanetary Trajectories	These trajectories, representing the celestial mechanics of the solar system, impose requirements on the spacecraft: geometrical variations of interplanetary distances and angles, for example, mission duration from launch to arrival, direction of approach to the target planet.
Encounter Geometry	Again, celestial mechanics dictates that only certain near-planet paths may be followed by the spacecraft. Local velocities and planetocentric distances undergo time variations which are consequences of these rules of mechanics, and must be accommodated by the spacecraft.

Interplanetary Environment	The spacecraft must be designed to survive its exposure to the interplanetary environment.
Planetary Environment	The spacecraft design must be compatible with the environment expected near the target planet.
Requirements of the Science Payload	For the complement of instruments which make up the science payload the spacecraft must satisfy requirements of weight, power, look angles and scanning, temperature control, commands, data handling, and transmission. In addition, there are limitations on adverse environments which may be generated by the spacecraft.
Trajectory Accuracy	Not only must a particular trajectory or class of trajectories be accepted as nominal goals but the tolerance in meeting these goals is expressed.
Schedule	As a result of the interplanetary trajectory study, major schedule milestones may be abstracted and identified.
Mission Duration	The duration of the mission is similarly abstracted from the trajectory analysis, although influenced by the scientific objectives.
Probability of Success	A target reliability for the mission is a matter for the application of judgment.
Growth Capability	Again, judgment, particularly concerning the role of the mission in a program of similar missions, pertains to the required growth capability of the spacecraft design.

For the 1972 Jupiter flyby mission, these functional requirements are analyzed in detail in Section 2 in Volume 2.

5.1.2 Interpretation of Requirements

The functional requirements given above are those of the mission, and represent the influence of science objectives and payload, target planet, and the like. In some instances, it is desirable to interpret these requirements in the light of the spacecraft design constraints adopted in this study, and outlined in Section 2.2. In addition, the assumption of spin stabilization for the attitude control mode for the spacecraft design reported in Volume 2 warrants interpretation of some of the requirements. These imterpretations are reported in Section 3 of Volume 2.

5.1.3 Generation of Spacecraft System Concepts

The major system areas, i.e., those whose characteristics and consequences transcend subsystem boundaries, in which alternate approaches exist, are addressed. For the design approach pursued in Volume 2, these areas include such factors as the mechanical design of the large antenna, the program for midcourse guidance and midcourse trajectory corrections, the means of orienting a spin-stabilized spacecraft by open-loop precession, and many other system-wide subjects.

5.1.4 Comparison of Concepts and Selection

In the areas given above, having system-wide implications, the approach is embodied in the format of Section 4 of Volume 2, in which the requirements are analyzed, alternate approaches are described, and the possible choices are compared, noting the major advantages and disadvantages of each, and the chosen approach is identified. Where applicable, an analysis indicates the extent to which the selected approach meets the requirements. This comparison of concepts and selection is done individually for each of the major system areas in Section 4 of Volume 2. A composite description of the resulting spacecraft system design is given in Section 7 of Volume 2.

5.1.5 Refinement of Requirements to the Subsystem Level

The steps outlined above comprise the first iteration of the spacecraft system conceptual design. Before it is possible to indicate the total feasibility, or to reiterate for improvement of the design, it is necessary to examine the implications of this design at the subsystem level. Therefore the functional requirements may be interpreted and refined to be stated at the subsystem level, in the light of the generated system design.

5.1.6 Affirmation of Subsystem Feasibility

From the refined requirements, stated at the subsystem level, a gross affirmation of subsystem feasibility is possible. To the extent that this affirmation cannot be established, grounds exist for alternation of the next spacecraft design concept iteration.

5.1.7 Iteration and Refinement

The steps described above are repeated, during which problem areas brought to light in the analyses of spacecraft system areas and in the affirmation of subsystem feasibility of the preceding iteration are accommodated. (One must concede that the possibility could exist that no number of iterations would resolve either system or subsystem problem areas. If such a result of the iterative procedure occurred, one would have to conclude that the mission objectives were unrealizable by the available technology or that they were not compatible with the study constraints and assumptions.)

Having satisfied that this process of iteration and refinement has selected a feasible design approach capable of implementation at the subsystem level, the selected design concept may be evaluated and further described. An evaluation against the original science objectives is given in Section 7.6 of Volume 2. The compatibility of the spacecraft design with applicable launch vehicles and the indicated choice of launch vehicle are discussed in Section 7.7 of Volume 2. Subsidiary discussions of the composite spacecraft design, consisting of a reliability analysis, cost effectiveness analysis, schedule, and cost estimate are given in Sections 8 through 12 of Volume 2.

5.1.8 Subsystem Design

The gross affirmation of subsystem feasibility is affected by the generation of alternate concepts, the comparison of these alternates, and a selection of the proposed concept at the subsystem level. Each of these processes is a miniature iteration of its own. The results of these iterations comprise the conceptual design of the subsystem of the spacecraft. A description of these subsystems, and the reasoning leading to the selection, is given in Section 8 of Volume 2.

5.2 THREE-AXIS STABILIZED SPACECRAFT FOR THE BASIC MISSION

Section 1 of Volume 3 is devoted to the spacecraft design based on 3-axis stabilization for the attitude control mode. The mission for this spacecraft is identical to the basic mission of the spin-stabilized spacecraft reported in Volume 2 and generated by the method described in the preceding section. The basic approach is to employ the same iterative method of responding to the mission functional requirements as outlined for the spin-stabilized spacecraft. Since the mission is the same in each case, the functional requirements are essentially identical.

The interpretation of these functional requirements is somewhat different, however, in that a 3-axis stabilized spacecraft accommodates some aspects of the requirements, for example, the experiment look angle and scan requirements, in an inherently different manner from the spinner. However, the major deviation enters the process in the generation of the spacecraft system concept, in which the treatment of the major system areas (analogous to the discussion in Section 5. 1. 3 above, for the spinner) necessarily entails different alternate implementations, and different bases for selection among the alternates. Of course, not all system areas are affected by the change of attitude control mode, and not all subsystems are affected.

Thus, the iterative process leading to the conception of the 3-axis spacecraft design, pursuing a path analogous to that of the spin-stabilized spacecraft, parallels it and terminates in identical or equivalent results in many areas. The description of the processes and the results (given in Section 1, Volume 3) does not repeat all the similarities, but concentrates on the major differences. However, the format in describing the 3-axis spacecraft design is the same as that used in Volume 2.

5. 3 MISSION VARIATIONS

The mission variations studied within the scope of the Advanced Planetary Probe study include variations in science payload weight, the year of the launch opportunity, the selection of the target planet, and whether the mission is a flyby, orbiter, or capsule entry mission.

The functional requirements which must be met by the spacecraft system design are sensitive to the mission definition, as well as to other influences. Table 1 indicates the major and minor sources of influence on each of the functional requirements. Because of this dependence of the requirements on the mission definition, the application of the iterative approach to the generation of a spacecraft conceptual design must recognize a changed set of functional requirements for each mission considered.

Table 1. The Sources of Functional Requirements of the Spacecraft Design

These functional requirements → arise from these sources.

These functional requirements	Study Constraints	Assumptions and Judgment	Solar System		Mission				
			Physics	Mechanics	Science Objectives & Payload	Class: Flyby or Orbiter	Year	Target Planet	Direct or Swingby
Launch vehicles	M								
Interplanetary trajectories	m	m		M	m	M	M	M	M
Encounter geometry		m		M	M	M	m	M	M
Interplanetary environment			M				m		
Planetary environment			M					M	
Requirements of the science payload					M				
Trajectory accuracy		m		m	M	M		M	M
Schedule							M	m	
Mission duration						M	m	M	M
Probability of success		M							
Growth capability		M							

M = major association

m = minor association

The nature of these mission influences on functional requirements is taken up in each section of the report, where appropriate.

For each mission variation, the flow chart of Figure 2 describes the process of generating the spacecraft design concept. As in the case of the 3-axis stabilized spacecraft for the basic mission, the mission variations are handled not by starting anew for the whole iterative process but by examining only those areas of the spacecraft design which must be altered to accommodate the new mission-sensitive functional requirements.

For the different mission variations, spin-stabilized and 3-axis spacecraft are considered. In some instances one concept or the other is selected for the basis of this study, with the grounds for this selection stated. In other instances a choice may not appear appropriate, and both concepts are retained for application to the varied mission.

6. SCIENTIFIC OBJECTIVES

6.1 JUPITER FLYBY MISSION

In this section the scientific objectives of the Jupiter flyby mission are identified, including the general types of experimental arrangements to be employed. In general, we have attempted to restrict the objectives to those measurements which require the presence of instruments in the interplanetary and Jovian environment itself. Those measurements which could be performed with balloons and rockets from the earth or from earth orbiters are not included. The experiments to be reviewed are listed in Tables 2 and 3.

6.1.1 Interplanetary Particles and Fields

The scientific objective of the interplanetary portion of this mission is to perform measurements which will provide information on the following questions:

- Where does the ordered solar wind flow pattern terminate? If the termination region is encountered, what is the magnetic field and particle configuration in this region?
- What is the magnetic field-particle configuration in the interplanetary medium at large distances from the sun? Are the distant measurements consistent with an extrapolation of near-earth values according to existing theoretical models? What temporal and spatial variations occur in the magnetic field and particle fluxes?
- How far do solar cosmic rays propagate in the interplanetary medium? How do their trajectories depend on distance from the sun?
- Does the Forbush decrease in galactic cosmic ray intensity extend to large distances from the sun?
- Does the intensity of galactic cosmic rays increase with distance from the sun?
- How do local measurements of particle fluxes and magnetic field correlate with solar disturbances?
- What is the spatial distribution of dust (micrometeoroids) in the interplanetary medium?

Table 2. 50-Pound Experiment Package in Spinning Spacecraft

Experiment			Sensor: No. Required	Sensor: Weight (Lb)	Sensor: Size (In.)	Sensor: Aperature Size (In.)	Sensor: View Angle	Pointing Direction	Electronics: Weight (Lb)	Electronics: Size (In.)	Electronics: Pwr (Watts)	Electronics: Data Rate (Bits/Sec)	Electronics: Commands*	Electronics: Diagnostic Telemetry
Interplanetary Particles and Fields	Cosmic Ray	A. Solar	2	1.5x2	(3x3x6)2	(3D)x2	$(60^{o})2$	1-Toward Sun 1-Away Sun	3	5x5x6	2	24	3	3T 1V
		B. Galactic	Sensor with Elect.			4.50	60^{o}	Away Sun	6	6x6x6	2	8	3	1T 1V
	Solar Plasma		Sensor with Elect			0.8x0.4	160^{o}x20^{o}	Toward Sun	5.5	7x6x6.5	1.5	24	3	1T 1V
	Radio Propagation		2	1.5	36& 6	-	-	Perpendicular to spin	5	7x7x6	1.5	8	2	1T 1V
	Micrometeoroid		Sensor with Elect.			5D	2π	Perpendicular to spin	4	6x6x6	1.0	14	2	1T 2V
	Magnetometer		1	1.5	2Dx3	-	-	1 Axis perpendicular to spin	4	6x6x6	4	24	4	2T 2V
Planetary Particles and Fields	Trapped Radiation	A. Electrons	Sensors with Elect.			1D	2π	Any	3	5x5x6	1	24	2	1T 1V
		B. Protons	Sensors with Elect.			1D	2π	Any	3	5x5x6	1	16	2	1T 1V
	Auroral		2	1x2	(3x3x6)2	2D	4^{o}	~Perpendicular to spin	3	4x4x5	2	32	2	3T 3V
Planetary Atmos. and "Surface"	TV		Sensor with Elect.			3D	4^{o}x4^{o}	~Perpendicular to spin	10	15x8x8	10	$1.5x10^{6}$	4	2T 2V
	Infrared Radiometer		Sensor with Elect.			2D	2^{o}	Perpendicular to spin	3	4x5x7	3	28	4	2T 2V
	Radio Occulation		PART OF SPACECRAFT SYSTEM											

*Including on-off.

Table 3. Characteristics of Experiments

Experiment	Measured Phenomena	Operational Principle	Operational Turn ON/OFF	Constraints	Noise
Cosmic Ray A. Solar	High energy particles >10 mev from sun	Analysis of single particles by suitable counter configuration	Turn ON at launch, OFF at $100R_J$	To see 360 deg in ecliptic plane	RTG Interference
Cosmic Ray B. Galactic	High energy particles >10 mev enetering solar system from stella medium	As above	Operate throughout mission	As above	As above
Solar Plasma	Angular distribution and flux of the solar wind protons and the nature of the solar wind/magnetosphere interaction	Curved plate electrostatic analyzer	As above	As above	-
Radio Propagation	Measure the integrated electron density between probe and earth	Measure phase shift between two propagated earth sent signals in non-vacuum medium	Turn on a launch use to limit of transmitted signal (if limited)	Antenna pointing and deploy constraint	Noise problem at these frequencies
Micrometeoroid	Properties and angular distribution of the interplanetary and planetary dust	Thin film-time of flight and microphone	Operate throughcut mission	Do not place in wake of spacecraft	-
Magnetometer	Interplanetary magnetic fields, magnetospheric boundary and planetary fields.	Tri-axial fluxgate null type	Operate throughout mission	Spacecraft magnetic background field >1 gamma	-
Trapped Radiation A. Electrons	Energy and angular distribution of energetic electrons in the magnetosphere	Geiger-Mueller counters with suitable absorbers	Turn ON at $150R_J$	-	-
Trapped Radiation B. Protons	Energy and angular distribution of protons in magnetosphere	Scintillation counter with absorbers	As above	-	-
Auroral	Emission spectrum	P.M tubes and filters	Turn ON at 10 to $50R_J$	Pointing constraint	-
TV	Complete planet coverage resolution ~40KM at $3R_J$ ~3000KM at $100R_J$	Conventional vidicon	Turn ON at $100R_J$ In use until fly-by	Large telemetry and/or storage required	-
Infrared Radiometer	"Surface" and upper atmosphere or cloud temperature	Optical and thermistor barometers with moving filters	Turn ON at $100R_J$	Light and dark N-S scan of planet	Avoid RTG look angle
Radio Occulation	Measure the integrated ionospheric and atmospheric density in planet vicinity	Uses spacecraft downlink frequency	Turn ON at $100R_J$	-	Noise problem at this frequency

Our present knowledge of the interplanetary field-plasma configuration has been derived from the following measurements:

- Direct plasma and magnetic field measurements have been performed with earth orbiters (IMP and OGO) and with Mariner 2, Mariner 4, Pioneer 5, and Pioneer 6
- Both terrestrial- and vehicle-borne studies of solar and galactic cosmic rays
- Studies of the deflection of Class I comet tails
- Studies of the Lyman α intensity in the night sky
- Studies of the Jovian nonthermal radio noise
- Radio propagation measurements

The direct plasma and magnetic field measurements have, in general, confirmed Parker's model (analogous to the pattern of a rotating sprinkler) for the solar wind within 1 AU but yield no information about the flow pattern at larger distances from the sun. Currently, this data indicates that the solar wind is always present with typical particle densities of 1 to 10/cm^3 and radially directed plasma velocities of 300 to 500 km/sec at about 1 AU. Accurate measurements of the electron thermal distribution have not been performed, but ion temperatures varying between 10^4 and 10^6 °K are observed. Typical values for the quiescent magnetic field are in the neighborhood of 5γ. Both the particle flux and magnetic field are enhanced during periods of geomagnetic storms. At 1 AU the interplanetary field angle is ~45 degrees. Extrapolation of Parker's pattern to larger distances from the sun yields a $1/d^2$ dependence in the solar wind flux and a 1/d dependence in the magnetic field intensity; the orientation of the lines of force is altered progressively from radial toward tangential with increasing distance. Thus, at Jupiter a solar wind flux of $2 \times 10^6/\text{cm}^2$ sec and a field of 1γ might be expected.

Although the momentum transfer mechanism between the solar wind and the comet tail plasma is not well understood and excessively high values of the solar wind flux have been derived from these observations, the study of tail deflections provides a useful monitor for the solar wind flow pattern, particularly in inaccessible regions. Observations of

Comet Humason indicate that the solar wind flow pattern is probably unchanged as far as the orbit of Jupiter.[1]

The integrated electron density between the earth and a spacecraft can also be obtained by radio propagation measurements, measurements of particular value because they provide data in the interplanetary plasma characteristics over large distances when contrasted with the local measurements performed with plasma probes.

Studies of Lyman α intensity in the night sky indicate that some 20 percent of the intensity is produced by energetic H atoms; the remaining 80 percent is attributed to cold H atoms in the terrestrial upper atmosphere.[2] A mechanism for the production of energetic neutral H atoms which traverse the interplanetary medium has been proposed by Patterson et al[3] based upon the termination of the solar wind as considered by Axford et al.[4] The latter propose that the solar wind, if low, is terminated by the interstellar magnetic field in a shocklike interaction at a distance where the solar wind kinetic energy density equals the interstellar magnetic field energy density. Depending upon assumed values for the solar wind velocity and density and the interstellar field intensity, termination should occur between 20 and 100 AU. Patterson et al[3] then proposed that cold neutral interstellar H atoms diffuse through the hot thermalized plasma in the shock region. A fraction of these atoms undergo charge exchange with energetic protons yielding an inwardly directed flux of energetic H atoms in addition to the cold atoms which penetrate the boundary. They show that the cold component is severely attenuated at 5 AU as a result of charge exchange with the solar wind and solar UV photoionization, but that the energetic component persists within 1 AU. These observations, therefore, provide evidence for the termination of the solar wind. A major objective of the proposed measurements would be a search for this transition region and an identification of the interaction details.

The Jovian radio noise observations indicate the presence of a radiation belt and a large planetary magnetic field. If the energization of the trapped particles is related to the incident solar wind flux as has been proposed for the earth, then these observations indicate that the solar wind reaches at least 5 AU.[5] The absence of radio noise from planets beyond Jupiter, with the possible exception of Saturn, may be related either to the absence of a planetary magnetic field or termination of the solar wind inside their orbits.

The studies of both solar and galactic cosmic rays also provide evidence for the plasma field configuration in the interplanetary medium. Spectra of both groups of particles show an exponential energy spectrum although the solar particles are considerable softer. Solar cosmic rays are copiously emitted in association with solar flares; there is some evidence that low fluxes may be continuously produced, and some production of energetic protons and electrons has been proposed to occur in the interplanetary medium.[6] Because some of the missions under consideration may occur in the early 1970's when solar activity should be maximum, these experiments are of great importance.

Terrestrial observations of the relationship between the solar flare location, terrestrial detection locations, and temporal behavior of these solar particles have yielded important information.[7] In general, the faster particles emitted from the western part of the sun tend to arrive at specific impact zones at the earth; for these cases the observed increase in intensity (rise time) is fast. Subsequently, the lower energy particles tend to arrive isotropically with a subsequent slow decay in intensity. The delay time between detection and the optical identification of the flare is longer than would correspond to transit of the straight line distance and is in good agreement with a particle path length following the spiral lines of force in the medium. For flares which occur on the eastern part of the sun, the fast rise time component is not observed and only the isotropic flux (with a slow rise time) is observed. These observations are consistent again with the magnetic field model which would preclude a direct path from the flare site to the earth. The subsequent isotropic arrival with durations long after the cessation of the optical flare and associated radio noise, would indicate that a storage and diffusion mechanism (this permits the particles to cross the lines of force) is probably active. A disordered although not necessarily chaotic magnetic field configuration may possibly provide the storage and diffusion mechanism. Some measurements suggest that this region can occur between Mars and

Jupiter. The observation of the propagation of solar cosmic rays, therefore, is an important aspect of interplanetary measurements.

Associated also with flare occurrences is a modulation of galactic cosmic rays. At a time following the flare, which corresponds to the arrival of the enhanced slow plasma component at the earth, a significant (30 percent) decrease in the flux of high energy galactic cosmic rays is observed. In the case of the earth, this effect occurs about 30 hours after the flare and after the solar cosmic ray intensity has decayed almost totally. This decrease, the Forbush decrease, is generally attributed to an enhanced interplanetary magnetic field which provides more effective shielding of the earth. An objective of these experiments is therefore an identification of the spatial extent over which the Forbush decrease is active.

A particularly interesting special case is the study of the propagation of solar cosmic rays produced by a second flare during the time the Forbush decrease is still present.[8] During solar cycle maxima such events can be anticipated. Measurements during such an event indicated an enhanced transmission of solar protons to the earth corresponding to a more efficient guiding by the magnetic field.

Typical cosmic ray measurements are therefore directed toward a study of the flux, energy distribution, mass and charge, and angular distribution of the particles and correlation of these observations with solar phenomena.

Observations of the zodiacal light and gegenschein provide evidence for the presence of dust or micrometeoroids in the interplanetary medium. These particles are believed to originate from the disintegration of comets or in the asteroid belt. The direct mapping of the distribution of dust in the interplanetary medium, particularly in the vicinity of the asteroid belt, represents an important aspect of any interplanetary missions.

6.1.1.1 Magnetometers

The purpose of the magnetometers carried on spacecraft is to measure the vector interplanetary magnetic field, the magnitude and configuration of the planetary magnetic fields, and fluctuations in magnitude and direction of the interplanetary and planetary fields.

Two types of magnetometers are proposed for this mission: the triaxial fluxgate and the triaxial helium. Both have been flown before, the helium on the Mariner 3 and 4 and the fluxgate type on Mariner 2, IMP, and Pioneer 6, although on the last two spacecraft the sensor was a single axis type.[9]

To cover the magnetic field level extremes, from 1 gamma for the interplanetary medium to 0.5 gauss expected during the Jupiter flyby, the magnetometer experiment has to have a dynamic range far greater than any magnetometer experiment flown to date. To detect the small changes expected during interplanetary flight periods, sensitivities in the order of 0.25 gamma would seem reasonable and well within the limits of present day magnetometers; for the magnetometer to make use of a sensitivity of 0.25 gamma, it is necessary that the magnetic noise level of the spacecraft itself be below this level at the magnetometer sensor. To cover the wide dynamic range of from 0.25 gamma to 0.5 gauss within the confines of an 8-bit digital word readout requires many switched ranges; each range being changed either by a ground command, automatic onboard sequencer, or both.

It is proposed that these ranges will not be linear in their coverage; e.g., the most sensitive range might cover from ±0.25 gamma through to ± 32 gamma with each digital bit representing a 1/4 gamma, whereas on the last magnetometer scale or range each bit change would represent changes in 100 gamma.

An alternative to the one magnetometer covering a large dynamic range and requiring many switched ranges is two separate magnetometers, each suitably scaled so that one covers the very low field levels and the other the higher field levels. Where weight permits, it is therefore proposed to fly two magnetometers, the helium magnetometer to cover the lower interplanetary magnetic fields and the fluxgate for the high magnetic fields in the vicinity of the planet Jupiter. The choice of the helium sensor for low level magnetic fields of interplanetary space is dictated by its fixed bias offset, an offset that having been determined prior to flight, is unchanging during flight. Unlike the fluxgate sensor where the offset is not known (unless some flipping mechanism is introduced), the helium is well suited for the detection of the very low field levels of interplanetary flight.

6.1.1.2 Plasma Probe

The purpose of the plasma probe is to provide detailed information about the energy distribution, directional distribution, and spatial and temporal variation of the ion component (H^+, He^+, and He^{++}) of the solar wind. Measurements of the electron component are uncertain because of photoelectric charging of the spacecraft and are not included in the program; however, the same instrument is usable for electron measurements should they be desired. The measurements for this experiment include the flux in the plane of the ecliptic as a function of solar angle, and the angular dependence of the flux out of the plane of the ecliptic.

The curved plate electrostatic analyzer[10] appears to be best suited for this experiment and has been successfully flown on the Pioneer 6, OGO, and IMP. It offers advantages over the Faraday cup type of plasma probe,[11] including better angular and energy resolution and the possibility for a significant increase in sensitivity. Also, due to the far smaller windows required, thermal leak problems are minimized.

At present, the optimum design for this instrument shows a lower sensitivity limit of 10^5 particles cm^{-2} sec^{-1} with ion energies determined over the range 100 ev to 15 kev and the electron energy over the range 3 to 500 ev. This lower limit of sensitivity appears to be determined entirely by amplifier noise and is perhaps adequate for a Jupiter mission, but not for farther missions. However, a significant improvement can be achieved by using single particle counting techniques (secondary emission multipliers of either the dynode or channeltron type) which will reduce the lower limit to about that of cosmic ray background.

6.1.1.3 Radio Propagation

The purpose of a radio propagation experiment is to measure the average interplanetary electron density between the earth and the probe and the time variations of this quantity. In addition, on a spinning spacecraft, a relativistic effect due to the ordered motion of electrons in the solar wind can be indirectly detected by measuring the degree of elliptical polarization of the received radio signals.

This experiment has flown successfully on Pioneer 6 and consists of two coherent receivers operating at 49.8 and 423.3 Mc, respectively, along with two whip antennas. The receivers are mounted within the spacecraft body and the antennas are mounted on the spacecraft high gain antenna feed unit.

Coherent signals transmitted from earth are received by the spacecraft experiment, where the phase advance and the group delay of the two signals is compared on the spacecraft. In this way the integrated electron density and its time variation is determined. A one-cycle Δf between the low frequency signal and 2/17ths of the high frequency signal implies a change of 4×10^{14} electrons/meter^2.

6.1.1.4 Cosmic Ray Experiments

It is proposed to include two different cosmic ray telescopes on the advanced planetary probe. The first will cover the energy range 10 to 200 Mev/nucleon and will permit analysis of the energy, mass, charge, and angular distribution of cosmic rays in this energy range. This instrument will be adequate for the study of both solar and galactic particles. In addition, a second telescope covering the range 200 Mev and above will be employed to permit the study of galactic particles unambiguously if the postulated disordered region in the interplanetary medium is encountered.

A satisfactory cosmic ray telescope for the 10 to 200 Mev region has been described by Fan et al[12] and has been employed on IMP and Pioneer. It consists of two solid state detectors to yield dE/dX measurements and also to define the telescope solid angle, a CsI-Tl detector, for pulse height analysis, which is further surrounded by a plastic scintillator. The acceptance angle of the arrangement is about 60 degrees and two such detectors will be employed because of the dependence of the direction of the lines of force of the interplanetary magnetic field on distance from the sun. The engrance aperture is covered with an aluminized mylar film to eliminate the photoelectric response to the semiconductor detectors; at the same time the heat leak is also reduced.

The higher energy detectors have not as yet been flown in earth satellites or interplanetary probes. Because of the high particle energies, measurements of these particles are obtained at balloon altitudes for the near-earth environment. Such a telescope which employs a Cerenkov detector and dE/dX scintillator[13] appears suitable for this experiment. Only one detector of this type, mounted in the anti-solar direction is desired; the entrance aperture should again be about 60 degrees.

6.1.1.5 Micrometeoroid Experiment

The purpose of the micrometeoroid experiment is to measure the flux, momentum, energy, and spatial variation in the flux of the small particles to be found in interplanetary space and surrounding the planets. Typical velocities range from 4 to 73 km/sec. It is expected that the largest flux of particles will be detected with their velocities in the plane of the ecliptic.

The properties to be determined are mass, velocity, and flux. Also, knowing the sun and roll angle of the spacecraft, the heliocentric orbit of the detected particle can be determined.

Proposed for this spacecraft is an instrument of a type flown on OGO 1 spacecraft and to be flown on the Pioneer C and D. The surface area, detector system solid angle, and sensor sensitivities determine the estimated rate of impact. This sensor is capable of detecting particles of mass greater than $5 \pm 0.5 \times 10^{-14}$ gram at velocities greater than 4 km/sec. Therefore, the sensor is capable of detecting dust particles near the desired mass threshold (10^{-15} gm). It is expected that the particle flux rate will range from 10 to 100 impacts per day and is based on a flux rate of 5×10^{-3} times that near earth, a detector area of 100 cm^2, and an acceptance angle of π steradians. The mass of the particles is expected to range from 10^{-11} to 10^{-15} gm and is based on the assumption that the particle density is 1 gm/cm^3 and a radius ranging from 0.1 to 2 micron.

6.1.2 Planetary Particles and Fields

The objective of measuring particles and fields in the vicinity of Jupiter is to provide information with respect to the following problems:

- What is the magnitude of the Jovian surface magnetic field? Is it dipolar? Where is the dipole located?

- Where is the magnetosphere-solar wind boundary? Are the characteristics of the interaction region similar to those of earth's transition region?

- What are the spatial distribution, energy distribution and fluxes of trapped particles (both protons and electrons) throughout the Jovian magnetosphere? Is the explanation of the decimeter noise as synchrotron radiation valid?

- Do auroras occur?

- Why does satellite passage influence decameter noise radiation?

- Are there significant magnetic anomalies, perhaps associated with the red spot?

- What are the characteristics of the upper ionosphere; the diurnal variations, variations with solar activity?

- Do plasma instabilities occur?

- What is the relationship between radio noise sources and energetic particle distribution?

If it is assumed that conditions in the interplanetary medium are unchanged at Jupiter and that the 50-gauss estimates of the Jovian surface magnetic field are reasonable correct, the sunlit Jovian magnetospheric boundary should occur at 50 to 150 R_J. In the terrestrial case this boundary is a shock-like region which may be several R_e in thickness. The inner boundary of the transition region appears to coincide with the boundary for durable particle trapping; energetic particles are also observed within the transition region and even beyond it, but it is clear that these are not trapped particles. The wider range in boundary location arises from uncertainties in the solar wind energy flux and the magnitude of the Jovian field.

Magnetometer and plasma probe measurements are required within the magnetosphere in order to establish the existence of the transition region. In addition, the magnitude and configuration of the Jovian field, and the inclination of the magnetic axis to the rotation axis (presently estimated as ~ 10 degrees from the radio noise observations) are desired. Therefore, both sets of measurements should be operative prior to the encounter with the boundary. The existence of any possible field anomalies which can be associated with observed features, such as the red spot, are most important.

Present information about the Jovian radiation belts is derived from nonthermal radio noise measurements.[14] The decimeter radiation is assigned to synchrotron radiation of relativistic electrons trapped in the planetary magnetic field. The decameter radiation is assigned to cyclotron radiation produced by lower energy (nonrelativistic but still energetic) electrons penetrating to low altitudes. The intensity of the planetary "surface" magnetic field is derived from this model. Also, an upper limit to the ionospheric electron density is set by the propagation of this radiation through the ionosphere. The radio noise measurements give no information whatsoever with respect to the proton energy distribution, but if it is at all analogous to the terrestrial situation, high energy protons may be expected.

Typical estimates of the Jovian radiation belt population indicate relativistic electron fluxes (5 to 10 Mev) about a factor of 10^3 greater than the terrestrial value, or about $10^6/\text{cm}^2$ sec. The fluxes of lower energy electrons are not clear.

The correlation of the intensity of the 10-cm noise with solar sun spot activity possibly indicates a correlation with the properties of the solar wind impinging upon the Jovian magnetosphere.[15] If the radiation belt population is maintained in a manner similar to that proposed by Nakada and Mead[16] for the terrestrial outer radiation belt, then the entire Jovian magnetosphere will contain accelerated suprathermal particles. If the population along any line of force is limited either by β $(8\pi N\ KT/B_2)$ or the whistler instability proposed for the earth by Kennel and Petschek,[17] then it may be expected that the energy distribution of the trapped particles will increase with increasing distance from the planet as $B/B_{boundary}$ where $B_{boundary}$ is the field intensity at the magnetospheric boundary.

Studies of the Jovian 10-meter radio noise suggest an upper limit to the peak ionospheric electron density of approximately $5 \times 10^6/\text{cm}^3$ if the low frequency cutoff is assigned to the ionosphere and the source is below the peak density. Therefore, a topside sounder would require a maximum frequency of 30 Mc; a reasonable sweep range is 3 to 30 Mc.

Optical searches from the earth for auroral phenomena on Jupiter have not been successful to date.[18] However, such studies are suggested by almost all groups considering future research programs for Jupiter. Present opinion is that the Jovian 10-meter radio noise is cyclotron radiation produced by moderate energy (nonrelativistic) electrons in the relatively strong, near-surface magnetic field. Such radiation, usually emitted in bursts, occurs almost every observing night and is therefore not an unlikely phenomenon.

The presence of moderate energy electrons at low altitudes is characteristic of terrestrial auroras, and as with the earth it is probable that in the case of Jupiter these are no longer trapped particles but are precipitated into the atmosphere. The presence of large fluxes of such particles in turn suggests that the particle lifetimes in these trapped particle belts is short and that the observed belt population represents an equilibrium population between efficient acceleration and precipitation processes. The identification of Jovian auroral phenomena would thus suggest that the dynamics of these radiation belts is similar to that of the earth's belts despite the different environmental conditions.

Secondly, the auroral processes permit the spectroscopic study of the constituents of the Jovian atmosphere and may yield information about the H/He ratio. In the case of terrestrial auroras, all H atomic light and He light arises from precipitating protons and α particles themselves which undergo charge exchange and charging reactions with the ambient atmosphere. Because of the negligible abundance of H_2 and He in the terrestrial atmosphere, all collisional excitation light is limited to O_2 and N_2. In the case of Jupiter, however, H_2 and He are expected to be the dominant atmospheric constituents and their excitation spectrum should be available. It is possible that minor constituents such as Ne might also be observed.

Dust particles may be gravitationally trapped to provide a dust cloud in the vicinity of the planets. In the case of the earth, the gegenschein is believed to be produced by scattering from such particles; the population in the anti-solar direction is enhanced as a result of solar radiation pressure to form a tail in a manner analogous to the behavior of the Class II or dust comet tails.

6.1.2.1 Trapped Radiation Experiment

The purpose of the trapped radiation experiment is to measure the energy, flux, and spatial distribution of the energetic electrons and protons trapped in the planet's magnetosphere. To measure this phenomena, we have chosen energy-insensitive detectors and selected absorbers to provide crude measurement of the energy distribution rather than determination of the energy spectrum from the output pulse height of an energy sensitive type detector. This choice is dictated primarily by reliability considerations, although it implies the use of multiple detectors.

For the electrons, it is proposed to use two Geiger-Mueller tubes suitably shielded, with window apertures of a size dictated by the particle flux across which will be placed absorbers. These will be selected to give the integral flux above 40 Kev and 1 Mev respectively. Such detectors have been used.[19]

Initially, these detectors have been placed within the spacecraft body since they have no scanning requirement, but it may be desirable to locate them apart from the spacecraft to reduce the contribution from X-rays produced by the interaction of energetic electrons with the spacecraft itself.

For the protons, we have selected zinc sulphide scintillation counters with absorbers to give the integral flux above 100 Kev and 1 Mev. Such detectors have been flown by Davis and Williamson[20] for measurements in the earth's magnetosphere. Zinc sulphide is a particularly attractive phosphor because it is almost completely insensitive to electrons and X-rays, and has a high efficiency for protons.

6.1.2.2 Magnetometers

The instruments to be employed will be the same as those used for interplanetary particles and fields.

6.1.2.3 Topside Sounder

The purpose of a topside sounder experiment is to determine the electron density profile as a function of altitude for altitudes above that at which the maximum electron density occurs. Measurements on both the sunlit and dark regions will give some data on the diurnal variations of ionospheric properties.

In this experiment a radio signal is transmitted from the spacecraft at a given frequency and reflected from the planet's ionosphere at an altitude where $f = 8.97 \times 10^3 \sqrt{n}$. Measurement of the time delay between the transmitted signal and detection of the reflected signal provides a measure of the distance from the spacecraft to the reflecting layer. With approximate knowledge of the spacecraft trajectory, an altitude profile can be derived. For an electron density of 10^7, the critical frequency is 27 Mc; thus, to cover a reasonable range in density, measurements should be performed over the frequency range of 3 to 50 Mc.

The power requirements for this measurement are large when compared to those required by the topside sounder employed on the Alouette satellite[21] because of the large distances to the critical altitude imposed by a flyby at 3 R_J. However, together with the radio occulation measurement, this represents the best technique for the determination of ionospheric characteristics from a remote flyby vehicle or orbiter. The possible implications of the decameter noise radiation on these measurements has not been evaluated. The sporadic nature of these noise emissions suggests the existence of quiet periods when these measurements will not be seriously perturbed, however.

6.1.2.4 Auroral and Nightglow Experiment

The purpose of measurements of auroral and nightglow is, first to determine the existence of energetic particle precipitation into the atmosphere (auroras), and second to derive some information with respect to the constituents of the Jovian atmosphere (H_2 and He) from a study of the dark side emission spectrum. The auroral zones are determined by the magnetic field configuration; since the present data indicates only about a 10 degree displacement for the magnetic axis with respect to the rotation axis, north-south scans with a typical resolution of 10 to 20 degrees in latitude seem adequate to identify the existence of auroral phenomena.

The simplest instrument adequate for both purposes appears to be a three-channel photometer arrangement similar to that proposed for Mariner 3. We have selected the visible region of the spectrum for measurement because of the difficulties associated with photometry of HeI lines in the far ultraviolet. The specific wavelength regions are $H\alpha$ or H_B (λ = 6565, 4861), HeI (λ = 5875 or 4921) and an NeI line in the visible.

A ultraviolet spectrometer together with a photometer for HeI (λ = 5875 or 4921) would provide additional information about the atmospheric constituents. Such an instrument is described by Corliss[22]. With this instrument which scans the wavelength range λ = 875 to 3200 A, it should be possible to detect H, O, A, N, C, and perhaps other trace elements. In the instruments developed to date, typical scanning rates of the spectrum are 24 A/sec in first order and 12 A/sec in second order. The requirement of a spectral scan and the apparent large weight of these instruments would appear to limit their use to the large attitude stabilized vehicle.

6.1.3 Planetary Atmospheres

Here information is desired on the following questions:

- Is Jupiter composed of primordial solar material?
- What are the atmospheric constituents? Has Jupiter lost hydrogen during its evolution? What is the hydrogen-helium abundance ratio?
- What is the atmospheric scale height? What is the temperature distribution in the upper atmosphere?
- Is the Spinrad effect, which indicates a differential rotation for different molecular species, real?
- Are there electrical disturbances in the lower atmosphere?
- Is the lower atmosphere turbulent? Are there convection currents and winds which might indicate an internal heat source?

- What are the more detailed characteristics of some observed features such as the red spot, bands, and transient spots?

Consistent with Öpik,[23] the discussion of the Jovian atmosphere is restricted to the region above the cloud layer since the Jovian surface has not been well defined. The primary areas of interest include the atmospheric scale height and the atomic and molecular abundances. Current estimates of the scale height have been derived from a single observation of the fading of starlight from σ Arietis during an occulation by Jupiter. This measurement indicated an approximate scale height of 8 km; with an assumed temperature derived from μ wave and infrared observations of 130°K, this yields a mean molecular weight of about 4 for the Jovian atmosphere at altitudes just above the clouds. Based on this observation, and the observed NH_3 and CH_4 abundance, Öpik suggests the following composition for the Jovian atmosphere:

Molecule	He	H_2	Ne	CH_4	A	NH_3
%	97.2	2.3	0.39	0.063	0.042	0.0029

The proposed large abundance of He relative to H_2 requires a detailed explanation. Because of the large mass and low temperature of Jupiter, thermal escape of the low mass gases is almost totally inhibited. It is expected that Jupiter is far more representative of the primordial material from which the planet was formed than is the earth. The cosmic abundance of H_2 is a factor of 10 > He and, therefore, this proposed atmosphere requires the existence of processes which permit the escape of H_2. Theoretical arguments for such processes have been advanced independently by both Vrey[24] and Öpik. Clearly, the direct measurement of the He/H_2 ratio represents an important measurement in understanding the evolution and current structure of the planet.

For similar reasons, several important isotopic abundance ratios such as D/H, A^{36}/A^{40}, $Ne^{20}/Ne^{21}/Ne^{22}$ present at Jupiter are expected to be more representative of primordial material than those currently determined terrestrially. In the Fowler, Greenstein and Hoyle[25] model for the evolution of the solar system, small planetessimals which later formed the planets were, at one time, irradiated with intense fluxes of energetic solar protons. The resultant neutrons were thermalized and captured to yield an enhanced abundance of the neutron rich elements and isotopes such as D and K^{40} which decays to A^{40}. Because of the greater distance from the sun, such processes would be less important at Jupiter and a depletion in neutron rich isotope probable. Therefore, the measurement of specific isotope ratios would be of importance in our understanding of the origin of the solar system.

Emission spectroscopy of the light produced by auroral displays represents an attractive method for the identification of specific elements (H_2, He, Ne) and estimates of their abundances. These measurements will not yield isotopic abundance ratios and unfortunately those measurements must be postponed until direct mass analysis is possible.

VLF electromagnetic waves may be generated by electrical discharges in the lower atmosphere, such as lightning, and perhaps other sources including auroras. The ordinary component of these waves is propagated along the lines of force of the magnetic field with small attenuation for frequencies less than the minimum electron-cyclotron frequency along the path provided a sufficient ambient electron density is present. Because of the time dispersion in received frequency, these waves are called "whistlers".

The study of whistlers has been an important method employed in deriving information about the terrestrial upper ionosphere out to 7 earth radii including the absolute density dependence on planetary distance and temporal variations in this distribution. In the case of Jupiter, however, the identification of these waves, in itself, is an important objective since it would be indicative of the existence of low altitude electrical disturbances.[26] The derivation of detailed ionospheric properties would await later measurements.

Atmospheric scale heights can be obtained with either optical or radio propagation techniques. The former detects the neutral atmosphere, the latter both the ionosphere and neutral atmosphere. Occultation is, of course, required for the radio propagation experiment in order to determine a surface atmospheric density. However, the entire model for the experiment is invalid at the expected "surface" densities expected for Jupiter. There will be some altitude, with an associated density, during a flyby experiment for which the experiment is valid. This, of course, implies that occultation is not necessarily important for this experiment.

The radio propagation technique has been employed for both measurements of the interplanetary electron density and on the Mariner Mars flight to obtain both the Martian surface density and neutral and ionospheric scale heights. There are reasons why these measurements may yield incorrect values, based mainly upon the theoretical assumption of a quiescent plasma.[27] The possible presence of density fluctuations in the planetary ionosphere can be determined with a fluctuating electric field detector as flown by Scarf et al[28] and are of great importance in the evaluation of the radio propagation experiment.

The possibility of obtaining high resolution TV images of the planet Jupiter raises a variety of important questions related to the circulation of the atmosphere. The existence of a large-scale circulation regime in Jupiter is known from ground observations. The pattern of bands and belts observed suggests a meridional cellular pattern, which is continuously varying down to the limits of resolution that can be reached from ground, about 100 km. The variations take place both in a time scale of the order of one Jovian day, and also over a long scale probably related to the solar cycle. The short time scale variations of the cloud structures and storm systems would be expected to reveal increasing detail when the resolution is increased, say to 10 km. Just as it is done in ground telescopes, by means of color filters, various levels of the atmosphere could be isolated to study in further detail the variations in depth of the cloud patterns. It should be noted that the atmospheric circulation patterns will yield information about the possible existence of an internal heat source. However, interpretation of these patterns will be dependent upon the detailed temperature measurements to be performed concurrently.

6.1.3.1 Microwave Radiometer Experiment

The purpose of a microwave radiometer experiment is to measure the Jovian upper atmosphere and "surface" temperature in the 1 to 2 cm bands by detection of the radiated microwave or electromagnetic flux.

Dictating the selection of these bands is the high noise level at the 10 cm frequency in the Jovian vicinity. It is intended that both the light and dark sides will be scanned in a north-south direction during the flyby.

An instrument similar to that flown on Mariner 2 will be used,[29] consisting of a parabolic dish for the detection of the microwave radiation, along with horn-type antennas for cold-space calibration. The field of view will be in the order of 2 degrees, giving a resolution of approximately 7000 km.

6.1.3.2 Infrared Radiometer Experiment

The purpose of the IR radiometer experiment is to measure the "surface" and upper atmospheric temperatures of the planet Jupiter. It is intended that this instrument will scan the light and dark sides of the planet in a north-south scan during the flyby period. The simplest instrument is an infrared radiometer like that of the Mariner 2. However, in the case of Jupiter, it is necessary to change the wavelength regions to be studied from that of Venus. Here we are interested in the 9 to 13 micron bands which is the absorption band of ammonia, and 1 micron for the absorption band for methane, and 2 bands, one below 1 micron and the other somewhere between 1 and 10 microns, which would give measurements more characteristic of the lower atmosphere or "surface."

This instrument consists of the optics through which the infrared radiation is focused, followed by a chopping mechanism which alternately exposes the detector to the planet and for reference purposes, free space. The beam is then split by a dichroic filter. The detector consists of two uncooled thermistor bolometers immersed in germanium lenses. Interference filters define the radiometer channels. The selected field of view for this experiment is 2 degrees giving a resolution in the order of 7000 km.

6.1.3.3 Radio Occultation Experiment

The purpose of a radio occultation experiment is to measure the scale heights of the Jovian atmosphere and ionosphere, and also provide a valuable supplement to the other onboard experiments used in the determination of such properties as composition, temperature, and density.

The simplest occultation experiment placing the least demands on the spacecraft system as a whole utilizes the spacecraft downlink signal. The received signal is detected and phase compared on the ground. Phase shifts in opposite directions are produced when the radiation is transmitted through an ionosphere and through a neutral atmosphere. Since the dominance of each effect is dependent on altitude, both requirements may be studied in the same experiment. This form of experiment has provided the most reliable measurement of surface pressure and scale height of the Martian atmosphere.

In the case of Jupiter there is no formal surface, so the surface pressure cannot be measured; furthermore, the entire model breaks down at the very high neutral pressures to be expected below cloud altitude. There is, therefore, no very formally prescribed trajectory for this flyby mission.

As has been mentioned, these measurements are only valid for a quiescent ionosphere. The existence of ionospheric instabilities can be determined with an AC electrometer. In this experiment the capacitively-induced voltage difference between the spacecraft and a short antenna produced by short wavelength electrostatic waves is determined. The antenna employed for this experiment could be the whip antenna of the two-frequency interplanetary radio propagation experiment.

6.1.3.4 VLF Experiment

The purpose of a VLF experiment is to measure the "whistlers," the existence of which will yield information about the currents of low altitude electrical disturbances. The best detector to employ for this is a simple search coil, consisting of a 50-turn loop antenna, 12 inches in diameter. Since we are interested in frequencies above 1 cps, when used on the spinning spacecraft, spin modulation is unimportant. A frequency range of 700 cps to 10 kc should be adequate for this instrument. Such an instrument[30] has been flown on Injun 3.

6.1.3.5 Television

The television camera design chosen for a Jupiter flyby mission is strongly affected by the stability mode and payload weight allocation of the spacecraft. The following requirements, however, are common to all missions:

- The best surface resolution obtained should significantly improve that available under optimum earth-based conditions. A resolution of better than 100 km at the planet is a reasonable goal.

- The camera must have the capability of pointing at the planet as the attitude of the spacecraft relative to the planet changes during encounter.

- The camera must have sufficient signal-to-noise ratio at the average planet brightness of 160 foot-lamberts and accommodate a wide dynamic range, perhaps 64:1.

- The camera readout mode must be consistent with the data storage and transmission limitations of the spacecraft.

- The camera must have sufficient environmental resistance to survive the total mission environment, including launch vibration and encounter radiation levels, with a high probability of successful completion.

A detailed description is given at the end of this section of a camera design which was selected for the 50-pound spin stabilized craft. Table 4 indicates changes which would occur in the basic system if adapted to other mission characteristics.

The camera design which was considered optimum in light of the considerations above consists of a single-axis tracking mirror, an f/3.3 refractive optical system, a shuttered SEC vidicon camera tube, and an auxiliary planet sensor sharing the main optical system. The operating sequence planned for the camera involves initial target acquisition using the planet sensor, long frame time pictures from 100 radii transmitted line by line making use of the long term storage capability of the SEC tube and short frame time, high resolution pictures taken near encounter and stored for future transmission. A selection of spectral filters for multi-band photography will be desirable only in the case of heavier or 3-axis stabilized probes.

Table 4. Comparison of Television Weight and Resolution for Four Spacecraft Configurations

Stability Mode	Camera Weight (lb)	Resolution at $1.5R_J$ (km)	Resolution at $100R_J$ (km)
Spin	10	15	1000
Spin	20	9	600
3-axis	10	9	600
3-axis	20	4	270

6.1.4 Observations of the Planetary Surfaces

The important problems to be studied include:

- Does Jupiter have an internal heat source? What is the relationship between the temperature of the illuminated and dark faces?
- Are temperature variations associated with any of the observed features?

The existence of a Jovian surface is questionable. All observed visible features are limited to the cloud layer, although the observed patterns may be determined by the underlying surface features. It is doubtful whether a flyby mission can yield any significant information about either the "surface" or the interior, with the exception of "surface" temperature measurements. On the other hand, the identification of an internal heat source together with the characteristics of the magnetic field will yield important information with respect to the planetary interior.

There is some evidence to support the conclusion that an additional heat source to solar radiation is present in the Jovian interior. This heat source may be nuclear, chemical, gravitational, or rotational in origin. Present terrestrial observations are limited to the 9 to 13 micron infrared region and the 1 cm micron wave region and because of the absorption characteristics of NH_3, are limited to cloud altitude. The extension of the infrared measurements to longer wavelengths is most important in the establishment of surface temperature; the micron wave measurements are to a degree redundant. A comparison of sunlit and dark temperatures, of course, requires a flyby mission.

Temperature mapping in both the infrared and micron wave regions is most important to determine whether temperature variations are associated with any of the visually observed features. The spatial resolution achieved with a near approach is very important.

6.1.5 Objective Subsequent to Planetary Encounter

Except in the special case of a swingby mission, the objectives subsequent to encounter are identical with the interplanetary objectives discussed in Section 6.1.1. The instrumentation described in Section 6.1.1 should be adequate to the interstellar boundary.

6.1.6 Spinning Versus 3-Axis Control Spacecraft

The scientific instruments impose many constraints on the overall spacecraft design, ranging from weight and size to the communications system and power requirements. The choice of experiements has been made to cover the range of phenomena detailed in Section 6.1.1 through 6.1.4. The means of dispatching the spacecraft to the planets, the time of flight and accuracy of the trajectory, and the necessary pointing accuracy of the downlink communications antenna are discussed in Volumes 2 and 3. Here we will discuss the limitations imposed on the spacecraft by the scientific payload, in view of whether the spacecraft is spinning or stabilized in three axes.

6.1.6.1 Spinning Spacecraft

During the interplanetary stage of the mission, it is required that measurements be made of the solar wind, the interplanetary magnetic field, the solar cosmic rays, dust distribution, integrated electron density between earth and the probe, and finally, the galactic cosmic rays. For the solar plasma and solar cosmic ray experiments it is required that the sensor be positioned on the spacecraft such that during one spin cycle large angles are scanned which include the sun. On a spinning spacecraft utilizing an earth-oriented antenna, some difficulty arises due to the large (initially) and changing spacecraft-sun angle, requiring sensors with wide look angles or several sensors to cover the total angle with correspondingly narrow look angles. To accommodate these requirements, a plasma instrument has been chosen such that the sensor itself sweeps out an angle of 160 degrees and coupled with the spacecraft rotation an almost 2π field of view is obtained. The same requirements on the spacecraft are made by the solar cosmic ray experiment, also requiring a sweep through angles that embrace the sun. In this case the sensor field of view is more restrained to something like 60 degrees and a difficulty arises in looking at the sun the whole time due to its changing angle in relation to the spacecraft earth-pointing axis. To overcome this, two sensors are employed, and as with the plasma probe, the spacecraft rotation is utilized to sweep out large angles.

The micrometeoroid detector requires a position on the spacecraft so that its detection surface has a reasonably clear 2π view and pointing in the direction of most frequent impact. Since most micrometeoroids will impact the dark side of the spacecraft (i.e., most particles in the asteroid), this sensor is mounted there.

The remaining two experiments used during the interplanetary stage of the mission impose little, if any, restrictions on the spinning spacecraft, other than the placement of the antennas in a position to receive earth-sent signals in the case of the propagation experiment, and positioning of the galactic cosmic ray sensor on the spacecraft so that at all times it is directed away from the sun in the direction of flight with an unobstructed 60 degree field of view for the sensor. By angling the sensor in relation to the spin axis, one will sweep out a larger area if required, although this is not thought necessary.

During the planetary flyby stage of the mission, good use is made of a spinning spacecraft and fixed sensors to observe the surrounding ionosphere and atmosphere along with the planetary "surface" features, although, in the case of the TV experiment, a penalty is paid due to smearing of the picture. Also, with this experiment, due to its narrow field of view, some provisions have to be made to either change the angle of the camera as the spacecraft approaches the planet or use some changing optical means to reflect the planet's image onto the camera lens. The infrared radiometer and the auroral experiments have pointing requirements in the direction of the planet and are helped in their scanning requirements by the rotation of the spacecraft. The trapped radiation experiment does not require any scanning, and being omnidirectional, can be positioned almost anywhere.

6.1.6.2 Three Axis Controlled

As just mentioned, the majority of the instruments make use on a spinning spacecraft of this rotation to either "sweep out" an area in space, or when passing a planet to scan the planet in a north-south direction. This movement, along with the forward motion of the spacecraft, enables the whole planet to be covered. When adapting experiments to a 3-axis stabilized spacecraft, if the instruments are to cover and detect the same range of phenomena, changes have to be made. In the case of the plasma and solar cosmic ray experiments, there is no "cone-like" effect as that of the spinner; therefore, if the sensors are fixed, more will be necessary to cover the required angles or the individual experiments will have to be rotated, or, of course, a common scanning platform is required. If a common movable platform is used, other problems arise. It has to be of a size large enough to accommodate the TV camera, infrared, and auroral experiments (for the 50-pound payload. It also has to be positioned on the spacecraft so that it allows the experiments to see around any large fixtures such as the high-gain antenna. It should be required to scan through an angle of approximately 90 degrees in the plane of the ecliptic for the plasma and solar cosmic sensor, and through the same angle in the ecliptic as well as some smaller angles out of the ecliptic plane for the TV camera, infrared, and auroral experiments.

In favor of a stabilized spacecraft and a common scan platform, compared to a spinning spacecraft, are the following:

- The resolution of the TV experiment picture is improved by a factor of at least 2.

- The resolution of the infrared experiment is improved.

6.1.7 Television

The primary objective of a television camera experiment is to transmit to earth high resolution photographs of the Jovian surface. A secondary function is to correlate and monitor the visual appearance and location of the planet with other events during encounter. This section discusses possible approaches to achieving these objective and describes in detail a selected camera configuration. Particular emphasis is placed on the design for a lightweight spin stabilized probe since this is the most difficult case.

The desired angular resolution and planet brightness can be computed from readily available astronomical data. Under excellent seeing conditions, an earth-based telescope can resolve down to 0.3 $\widehat{\text{sec}}$ on Jupiter. At least earth-planet distance this resolution corresponds to approximately 1000 km. A television resolution corresponding to a 500-line picture with a four-degree field of view at a range of 1.5 Jupiter radii from the surface would resolve about 15 km, almost two orders of magnitude better than the earth based performance. A resolution of this order of magnitude is reasonable for the spinning probe. Better resolution is straightforward from a 3-axis stabilized platform.

The average planet brightness can be computed from the planetary magnitude and angular semi-diameter as measured from the earth. The magnitude of Jupiter is given by Allen[31] as -2.4 when observed at a range of 4.2 AU, equivalent to a semi-diameter of 2.3×10^{-4} radian. Using the figure given by Allen for the illumination of a zero magnitude star the brightness is given by:

$$B = \frac{2.43 \times 10^{-10} (2.51)^{-2.4}}{\pi/4 \, (2.3 \times 10^{-4})^2} \qquad \text{lumens/cm}^2\text{-sec}$$

After rationalizing and converting to square feet:

$$B' = 160 \text{ foot-lamberts.}$$

The television camera for the spin-stabilized Jupiter probe is strongly constrained by the spacecraft parameters and experiment objectives. The following is a list of primary requirements and constraints for which the system is to be designed:

Resolution	$\sim 1 \widehat{\text{min}}$
Spacecraft spin rate an earth line axis	5 rpm
Planet brightness (average)	160 foot-lamberts
Signal range	64:1
Approximate mission duration	2 years
Available data transmission capacity	400 bits/sec

Several mission variables, notably the geometry of the planetary encounter, affect the degree to which the experiment objectives will be achieved. An additional constraint not specified in the above list is the requirement that the television camera weight be kept small relative to the overall science payload figure of 50 lbs.

6.1.7.1 Requirements and Alternatives

To select an optimum television camera configuration it was necessary to consider several possible alternatives, basically differing with respect to the detector type. Each approach was analyzed in terms of the following considerations which result from the mission requirements:

- Depending upon the closest approach at encounter, the camera must have a net angular resolution in the order of one minute of arc
- Due to the angular rotation of the optical axis at rates up to 5 rpm (30 deg/sec) the elemental exposure time must be limited to 0.5 millisecond
- The camera system must have the capability of following the target planet in the orbital plane during encounter and initiating exposure at the desired roll angle

- The camera data readout mode must either be suitable for immediate transmission or conditioned in such a manner that it may be stored in the spacecraft data system for future transmission.

The camera configuration which has the strongest intuitive appeal for this application is one which is capable of using the spacecraft spin motion for imaging. Such a camera might consist of a single-axis deflectable mirror for centering as the earth-spacecraft-planet angle changes and a detector array or image dissector for scanning normal to the roll deflection of the line of sight. If an image dissector were to be used in this scheme the horizontal scan would be limited to approximately 10 elements by sensitivity considerations. In order to construct a rectangular picture, say of 500 x 500 elements, 50 scans would have to be overlapped using the planet tracking mirror to establish adjacent picture elements. These scans would be linear only when the earth-spacecraft-planet angle is 90 degrees, no scan would be generated when this angle is 180 degrees. The horizontal scan of such a system could be increased by replacing the image dissector with an array of detectors of resolution element size. If the horizontal array consisted of 50 detectors only 10 scans could be required. In this case, however, the data rate would increase to 600 kbits/sec, making the data handling problem one of significant magnitude. Figure 3 illustrates the basic approach.

A second basic approach is to utilize a storage type image camera, capable of photographing the entire scene in a single exposure. In order to prevent loss of resolution due to smear the exposure period must be confined to 0.5 millisecond or less by a shutter. The illumination level on the sensor photosurface depends upon the f-number of the camera optics, jointly determined by field of view and packaging requirements. For this application it is unlikely that this illumination will exceed 10 foot-candles for the average planet brightness of 160 foot-lamberts. The net exposure is then 0.005 foot-candle-second, at which level a signal to noise ratio of 32:1 is required. This sensitivity level is somewhat out of the range of the most sensitive conventional vidicons but well within the range of more sensitive storage camera tubes, such as the image orthicon, intensifier vidicon, and SEC (secondary emission conduction) vidicon.

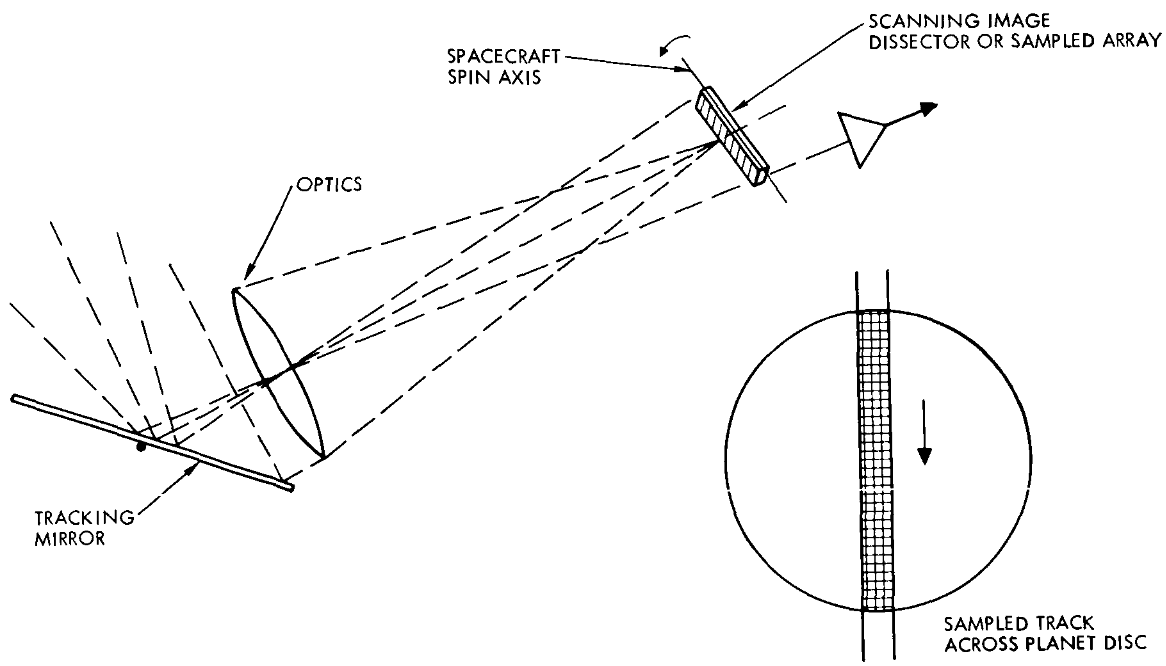

Figure 3. Spin Scanning Television Camera

Unlike the self-scanning camera system of Figure 3, the shuttered storage tube need not be read out during exposure. The storage surface of these tubes is capable of retaining the electronic image for a period varying from approximately one second for the conventional vidicon to an hour or more for the SEC vidicon. The latter capability leads to the possibility of direct transmission of each picture, eliminating the need for an in-line tape recorder at all times. The two approaches described above a self-scanning detector array or a shuttered image camera, appear to be the most straightforward approach to the television problem. Another approach of somewhat secondary interest but worthy of mention is a camera mechanization incorporating a "despinning" device. By introducing a motion of the camera optical axis in a direction counter to that caused by the spacecraft spin the exposure period permitted without excessive smear can be significantly extended. The exposure in this case might be extended to make possible the use of a conventional vidicon. The advantage gained in terms of camera tube size and simplicity would be opposed, however, by the difficulty of mechanizing the despinning. This motion can be generated by means of a rotating mirror in the optical path or by means of a special purpose image section of the camera tube.

In either case the mirror motion or image section deflection current must be carefully synchronized to the spacecraft spin cycle and be capable of variable speed to accommodate the somewhat uncertain spin rate. The various alternative camera approaches are summarized in Table 5 with a set of comparative parameters. Several existing cameras are included in this table.

In addition to the camera tube and its associated equipment there are several other significant elements in the television system, the planet sensor, shutter, filter asembly, and scanning mirror.

In order to expose the camera at the proper spin angle and to follow the planet in the orbital plane during encounter some tracking mechanism must be incorporated into the camera system. A fairly straightforward solution is to include a simple array of photodiodes in the focal plane of the camera and to process the signals corresponding to the crossing of the planet, or the planetary horizon, in a suitable tracking loop including the mirror drive and the shutter timing control. An alternate approach would be to design a dual mode camera capable of tracking the planet from the pick-up tube video in between commands to take a photograph. A third alternative is an entirely separate planet sensor unit.

The requirement to shutter the television camera for an exposure period of 0.5 millisecond is near the limit of performance for mechanical shutters. The best mechanical shuttering device is probably a traveling slit in the camera focal plane driven by a solenoid. If the total shutter time required to traverse the focal plane is 5 msec and the shutter dimension is 4 x 0.4 degrees, an average linear distortion of 2 percent will occur due to spin motion. The focal plane shutter offers the advantage of access to the optical path ahead of the shutter for a planet sensor beam splitter. An alternative shuttering scheme is gating of the image section in the case of the SEC vidicon. It is feasible to accomplish this electronic shuttering by switching the 8-kv image section potential. This rapid switching requires additional high voltage circuitry and risk of interference generation.

Due to the requirement for rapid exposure it is necessary to separate the filter assembly, if required, from the focal plane shutter

Table 5. Television Camera Characteristics

Detector	Device	Threshold Exposure (ft-cm-sec)	Resolution (TV lines)	Frame Time (sec)	Field of View (deg)	Unit Weight lb	Input Power (watts)	Comment
SEC vidicon	APP camera	3×10^{-5}	500 x 500	7200	4 x 4	10	10	Camera design proposed in this report
Image dissector	APP camera	1.3	500 x 500	10	4 x 4	10	8	Requires overlay of adjacent scans
Vidicon	APP camera	3×10^{-4}	500 x 500	3	4 x 4	8	8	Requires image motion compensation
Vidicon	Mars Mariner camera	7.5×10^{-3}	140 x 140	3	1.05 x 1.05	14	9	
Vidicon	Tiros camera	4×10^{-3}	550 x 700	200	-	13	NA	Operational Tiros
Vidicon	Ranger camera	4×10^{-3}	800 x 800	2.5	8.4 x 8.4	17	5.2	Ranger 9 F_B camera
SEC vidicon	Apollo camera	4×10^{-6}	220 x 290	0.1	-	4.5	NA	Continuous exposure hand held camera
Image orthicon	Stratoscope telescope	NA	350 x 400	.05	-	50	NA	Detects 12th magnitude stars

rather than combining the two functions. A solenoid-driven filter wheel in the optical path may be stepped to the next position between exposure times in a fixed sequence. Due to the narrow spectral response of most image tubes and the sensitivity limits imposed by the mission it may be that spectral filtering is not desirable.

The scanning mirror required to track the planet in the trajectory plane need be deflected at an extremely slow rate through a maximum angle of 45 degrees. Geometrical considerations indicate the desirability of mounting this mirror outside the spacecraft, driving it remotely from the main sensor, or through a small outboard drive mechanism. A flexure pivot mounted, magnetically driven device similar to that used on the Gemini and OGO earth sensors would be a desirable approach.

6.1.7.2 Selected Design

The camera system described here represents what appears to be the best combination of the system elements discussed above. It is not necessarily the optimum system for use in the 1970 era but it is a device which appears at this time to have a high probability of meeting the scientific objectives of the mission. The selected camera design consists of a single-axis scanning mirror, f/3.3 refractive optical system, SEC vidicon camera tube, focal plane shutter, planet sensor internal to the camera, and processing and logical electronics. No filter assembly has been included since it does not seem desirable at this time to do multi-band photography of Jupiter. A functional block diagram of the camera system is shown in Figure 4. The design and performance parameters of the system are as follows:

Camera

Weight	10 pounds
Power	10 watts
Optical system:	
Focal length	10 inches
Aperture diameter	3 inches
Transmission (including planet sensor beam splitter)	75 percent
Shutter	Focal plane, 0.5 msec

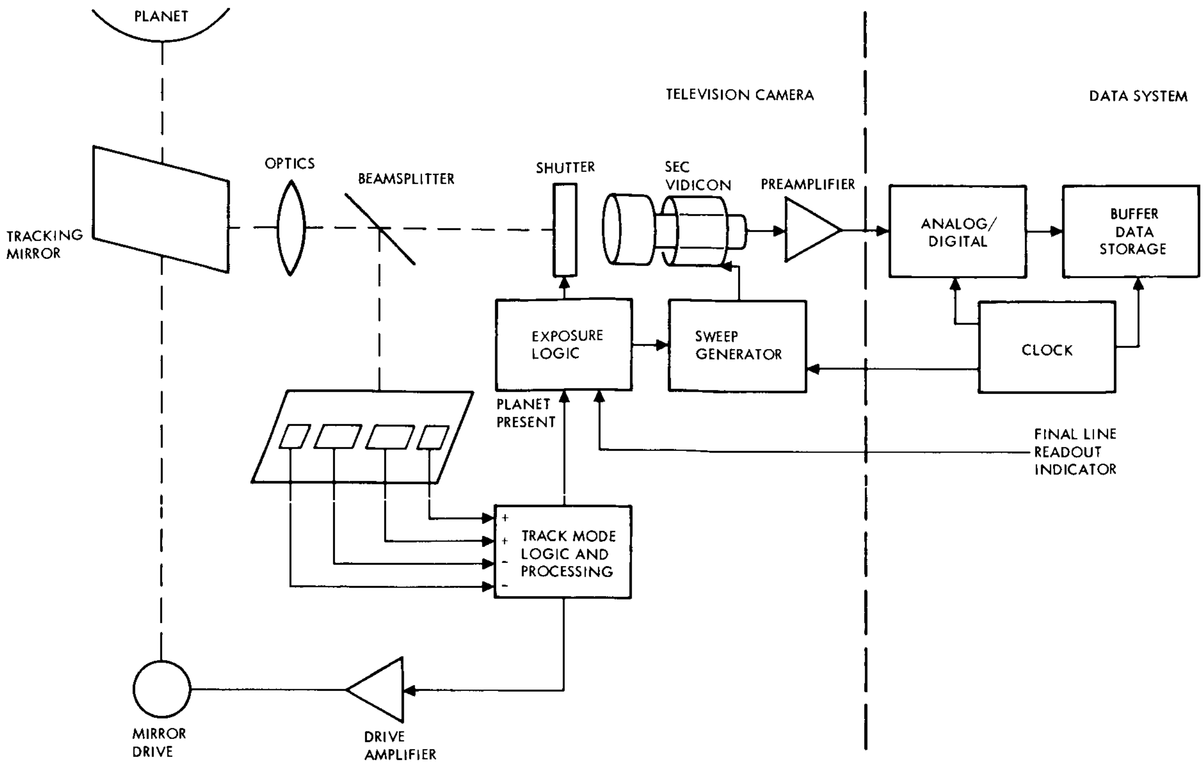

Figure 4. Camera System Block Diagram

Detector	
Type	SEC vidicon (similar to Westinghouse WX31003 but with S-20 response)
Spectral response	S-20
Horizontal resolution	500 TV lines (5 percent response)
Target format	0.70 x 0.70 inch
Power dissipation	2 watts
Planet Sensor	
Beam splitter	10 percent flux from main beam
Detectors	Dual arrays, silicon photodiodes
Function, mode 1	Acquisition, energy balance centering of "small' planet, exposure trigger
Function, mode 2	Centering of disc from horizon crossing, exposure trigger

As the Jupiter probe approaches the desired photographic range it is necessary to adjust the mirror angle such that the planet image (smaller than the field) passes through the center of the field of view during each spin cycle. This is done in two steps, first initiating a planet search pattern with the mirror until the large detector pair shown in Figure 4 indicates a planet crossing. At this time the mirror is driven through the differential processor and drive motor until the planet image is equally divided between members of the large pair as it passes through the field. This balanced condition is maintained during the phase of encounter for which the planet image is smaller than the field of view. A threshold circuit indicating the crossing of the planet provides a signal to trigger the shutter when a logic signal from the data system has enabled that device. When the planetary image exceeds the field size (at about 30 radii range) the planet tracker switches to a horizon sensitive mode, using the small outside detector pair. In this mode the mirror is driven to equalize the horizon crossing times experienced by the two detectors. This mode is sufficiently accurate to track the planet center up to encounter. It may be desirable to plan a programmed or ground commanded bias to reduce tracking error introduced by the apparent planet shape when significantly less than fully illuminated.

The camera system proper consists of the f/3.3 refractive optical system, shutter, SEC vidicon and its associated deflection components. The simultaneous existence of a planet present signal from the planet sensor and an enable signal from the data system causes the focal plane shutter to open and shut exposing the photocathode of the SEC vidicon to the planet image. The photo-electron image of the planet from the cathode is focussed and accelerated onto the secondary emission conduction and storage target of the tube. The electron beam scanning section of the tube then generates a standard video picture signal by scanning the storage target at a line frequency compatible with the data mode. The f/3.3 optics and 0.5-msec exposure period produce an exposure of 1.5×10^{-3} foot-candle-second on the image tube surface, sufficient to assure a 32:1 SNR in the readout signal. The scanning format for readout is of two types, continuous or line by line with a time delay between lines sufficient to transmit the single line data. The affect of the mission environment and some derivation of quoted parameters is given in the camera performance section.

6.1.7.3 Camera Performance

This section describes camera system performance. The major system element of concern in such an analysis is the sensor, with some attention required also to the scanning mirror, planet sensor, and shutter.

The illumination of the planet image on the tube photosurface is determined through the usual equation:

$$E = \frac{B\,T}{4F^2} \quad \text{foot-candles}$$

where

B = planet brightness, 160 foot-lamberts

T = transmission of camera optics, including loss to planet sensor = 0.75

F = camera "f-number" = 3.3

therefore

$$E = \frac{(160)\,(0.75)}{(4)\quad(3.3)^2} = 2.8 \text{ foot-candles}$$

and the integrated exposure over the 0.5 millisecond shutter period is $E\,\Delta t = 1.4 \times 10^{-3}$ foot-candle-sec. A reasonable responsitivity for an S-20 photoemissive target to radiation of solar color temperature is 150 microamps/lumen; therefore with a raster format of 0.70 x 0.70 inch the net charge in the planet image is

$$i\Delta t = \frac{(0.70)^2}{(12)^2}\;(1.4 \times 10^{-3})\;(1.5 \times 10^{-4})$$

$$i\Delta t = 7.15 \times 10^{-10} \quad \text{coulombs}$$

The net charge produced per resolution element in a 500 x 500 (N x N) resolution element scene is

$$\frac{i\Delta t}{N^2} = \frac{7.1.5 \times 10^{-10}}{(500)^2}$$

$$\frac{i\Delta t}{N^2} = 2.9 \times 10^{-15} \text{ coulomb/resolution element}$$

and the number of photoelectrons per element is

$$q_E = \frac{(i\Delta t/N^2)}{e}$$

where e is the charge on an electron, 1.6×10^{-19} coulomb:

$$q_E = \frac{2.9 \times 10^{-15}}{1.6 \times 10^{-19}}$$

$$q_E = 1.8 \times 10^4 \text{ photoelectrons/resolution element}$$

The inherent SNR is very nearly the square root of q_E or approximately 130. Westinghouse has indicated performance of the same order of magnitude as the inherent value when the degrading effects of the storage target, finite reading beam, and amplifying electronics are accounted for. Therefore it is felt to be feasible to predict a SNR of 32 or greater under the conditions of average planet illumination. Obtaining performance over a dynamic brightness range of 64:1 might require gain control via the image section accelerating potential.

The angular resolution capability of the television camera is jointly determined by the basic camera characteristics and the uncompensated motion of the camera axis during exposure. The basic camera resolution is determined almost entirely by the resolution of the detector since even a moderate quality optical system is capable of far better than one minute of arc. The selected detector has a horizontal resolution of 500 TV lines on the center of the photocathode (5 percent amplitude response). The angular motion of the camera optical axis varies from

0 to 30 deg/sec during encounter for a spacecraft spinning about the earth vector. The resulting "smear" in an 0.5 millisecond exposure period varies from 0 to 0.9 $\overset{\frown}{\text{min}}$ during encounter with the direction of the "smear" being parallel to the vertical raster sense in the maximum case and along a section of a circle of decreasing radius in other cases. Assuming an average value of smear, distributed equally in the two resolution "axes" and combining the two resolution degradations in terms of their equivalent aperture diameters,[32] the system average resolution is 1.1 min x 1.1 $\overset{\frown}{\text{min}}$. A more precise statement of resolution must await a definite trajectory choice and further indication of the SEC vidicon performance.

The planet sensor included in the optical path of the television camera serves the dual purpose of triggering the shutter at the proper spin angle and providing tracking signals for the gimballed mirror. For the particular sensor array shown in the previous section the most difficult tracking case occurs when the planet subtends an angle at the spacecraft large relative to the camera field of view. A brief analysis shows that a one-degree offset of the mirror angle in the orbital plane from the planet center will introduce a 6 msec shift in the horizon crossings detected by the two photodiodes, when the planet subtense is 40 degrees. This result indicates that control of the camera line of sight in the order of 1 degree accuracy is a reasonable requirement.

The previous discussion has been concentrated upon a television camera designed particularly for a lightweight, spin-stabilized spacecraft. In this section the effect upon the camera performance of extending the weight allocation or providing a completely stabilized platform is briefly considered.

One of the limitations of a small science payload is that the size of the television optical system and associated mechanical equipment must be constrained. If a 100-pound (or greater) payload is allowed, longer focal length optics with the proper "f-number" would become feasible. Under these conditions a 500 TV line system with a 2 or even 1 degree field of view might be desirable. Although the problems of acquisition and tracking would be somewhat more difficult, particularly in the spin-stabilized case, it is certain that a superior angular resolution would be obtained. In addition to the resolution advantages it would become more feasible to incorporate a system of optical filters into the heavier camera system.

Due to the "smear" introduced by spacecraft rotation, the spin-stabilized configuration is not as desirable from the television standpoint as is a three-axis stabilized craft. From a fully stabilized platform the television system would be faced only with the necessity of tracking the fairly low rate apparent motion of the planet in the orbital plane. The exposure or shutter period could be significantly extended, relieving the sensitivity constraint upon the detector and optics.

6.2 FLYBY MISSIONS TO PLANETS BEYOND JUPITER

Our knowledge of the planets beyond Jupiter is far less than our knowledge of Jupiter itself. There is, at present, no evidence supporting the existence of planetary magnetic fields and trapped radiation belts. In fact, it is not established that the solar wind extends to Saturn. Infrared temperatures and the abundances of NH_3 and CH_4 have been determined for Saturn but are only inferred for Neptune and Uranus.

6.2.1 Objectives

Studies of the outer planets require survey measurements to establish the existence of certain phenomena. Although in the case of Jupiter sufficient evidence exists to establish operating ranges for most experiments, much less is known of Saturn, Uranus, Neptune, and Pluto.

The important parameters to be determined include:

- Does the solar wind reach the planet?
- Does the planet possess a magnetic field? Is a magnetosphere present?
- Are there trapped radiation belts?
- What are the CH_4 and NH_3 abundances?
- What is the atmospheric scale height?
- What is the surface temperature?
- What are the characteristics of the ionosphere?

- What is the density and material of Saturn's rings?
- What is the difference in density between planets like Uranus - Neptune and Jupiter - Saturn?
- Can any inferences be drawn concerning the fact that the Uranus rotation axis is in the plane of the ecliptic. ?

6.2.2 Measurements

For missions to the outer planets, even if the flight time can be made relatively short, the basic precursor tasks will be to measure the interplanetary space. Therefore, the emphasis will be upon experiments which do this, such as magnetometers, plasma probes, galactic and solar cosmic rays, and radio propagation experiments. Because the sun's influence will be very small, the solar wind and the solar magnetic field will be greatly perturbed. These facts will make instrument sensitivity of critical importance and require a wide field of view, probably one which is variable. The long mission lifetime will set a premium on reliability.

With respect to the planetary experiments, it is extremely desirable that some type of picture be taken since the improvement in resolution over earth measurements for these planets is great. The data rate capability for any spacecraft to these planets will be very low; for the 50-pound payload spacecraft, a data rate of about 250 bits/sec at Saturn's orbit is possible and at 75 bits/sec at the Uranus orbit. While these bit rates imply the frequent ground station transmission at the assumed sample rate given for the Jupiter mission, the fact that most quantities are slowly varying will allow a considerably reduced sample rate. Although the TV experiment has a very high sample rate, the use of a tape recorder would appear to be very marginal for such long life missions. Therefore, an image storing technique with a real-time transmission is desirable. At 250 bits/sec, a picture can be transmitted once every two hours, which would permit a relatively large number of pictures. Only a few very high resolution pictures would be transmitted since there is little time to transmit during close approach.

At present no data exists indicating that there are trapped radiation belts at these planets but its seems reasonable to assume, using Jupiter as a model, that they would indeed have such belts. Detection of a possible aurora would not be appropriate until a magnetic field and radiation belt had been detected. An infrared radiometer would in general be a desirable experiment, if weight permitted, to measure NH_3 and CH_4 abundances. The radio occultation should be performed on each mission in order to detect a measurable atmosphere. It might also provide some interesting insight into Saturn's rings.

It is interesting to note that even a simple complement of experiments can solve many important scientific questions. A flyby of Venus and Neptune with a single picture could do much to resolve the questions of planetary masses and their density. A flyby of Uranus carrying only a magnetometer might well give us some insight into the anomaly that the axis of the rotation of Uranus is in the plane of the ecliptic.

Another way to look at this problem is to consider the Jupiter swingby missions. For such a mission the normal payload of planetary and interplanetary experiments would be carried to be used during the Jupiter passage. The same payload would then be carried on to succeeding planets. Although the payload would be designed specifically for Jupiter, as has already been indicated, it would be a suitable payload for the other planets.

6.3 JUPITER ORBITER MISSIONS

Once a spacecraft is in orbit about a planet, a great many experiments can be done in systematic fashion. Spacecraft like OGO or Nimbus could perform extremely useful tasks in an orbit about Jupiter. On the other hand, a large deboost capability consuming a substantial portion of the payload is required, which in turn limits the amount of payload which can be carried. The deboost requirement is sensitive to the particular orbit to be selected and can be made quite small for a highly elliptical orbit. If the orbit is extremely elliptic, then the period becomes very long and since most time is spent at apogee, observing conditions are not good. However, excellent samples of the radiation belts and magnetic field characteristics can be derived in an elliptical orbit. In addition, experiements such as the radio occultation experiment or an infrared radiometer can be performed frequently, thus enhancing confidence in the data.

However, experiments such as picture taking, which is the most desirable experiment from an orbiter, are quite sensitive to the orbit eccentricity. With an orbiter whose bit rate is about 500 bits/sec, a picture can be transmitted back to earth every hour and a half without relying on tape storage. If the orbit is highly eccentric, perhaps the most valuable data will be from perigee, which is of short duration, and consequently the number of high resolution images possible per unit time are limited. The period of a satellite with an eccentricity of 100 Jupiter radii is 44 days. If a shorter period is desired, then a greater percent of the total payload must be applied to the deboost propellant.

The tradeoff between orbit characteristics and science payload is not a simple one. Highly eccentric orbits around the earth such as those for the EGO (Eccentric Orbiting Geophysical Observatory) have considerable utility, but do not provide the type of data one would want on an early mission. On the first mission to a planet, even orbiters should provide gross data concerning the broad meteorological and topological features. If an experiment cannot provide these sources of data, it does not satisfy the principal aspect of human curiosity. On the other hand, a highly eccentric spacecraft orbit can provide a substantial amount of scientific data concerning the origin and evolution of the planets. This fundamental difference in objectives is the key problem in selecting the scientific payload for an orbiter mission and, in turn, selecting the type of orbit. If we assume that the primary interest in an early orbit mission is purely scientific, a lightweight orbiter carrying a modest payload and going into a highly eccentric orbit is acceptable. This type of objective seems to be the most appropriate mission to be considered.

6.3.1 Scientific Objectives

Objectives of a purely scientific Jupiter orbiter mission include the following.

- What are the temporal variations in the flux, energy distribution, and spatial distribution of the trapped protons and electrons. How are these variations correlated with solar activity? How are these variations correlated with the radio noise patterns detected on the earth?

- What are the temporal and spatial variations in the occurrence of auroral phenomena? How are the occurrences of aurora related to the trapped particle characteristics and solar activity?

- What is the magnitude of the Jovian surface magnetic field? Is it dipolar? Where is the dipole located?

- Why does satellite passage influence the decameter radiated noise? Does this occur because the possible magnetic field of the satellite perturbs the trapped particle confinement geometry?

- Is the Murray effect real? Does the surface temperature in a region shadowed by a satellite increase?

- What are the longer term variations in "surface" features? Here we imply time periods longer than the several days attained in a flyby mission. Is there any correlation between surface features and magnetic anomalies?

- What are the general temporal variations in all the characteristics enumerated in the flyby mission including ionospheric density, atmospheric and "surface" temperature, electrical disturbance, etc? How can these be correlated with terrestrial observations and solar activity?

It is possible that some information can be obtained with respect to the Jovian satellites including direct observation, identification of the existence of an atmosphere and some constituents, and the existence of satellite magnetic fields. These observations should form part of the experimental program.

6.3.2 Measurements

Because there are two basic orbits possible, a highly elliptical orbit whose objectives are purely scientific, or a near circular orbit whose objectives are not only scientific but of general interest, two types of payloads are possible. The first payload which can represent a large percentage of total spacecraft weight would include magnetometers, trapped radiation sensors, a radio occultation experiment, infrared radiometers, aurora detectors, visual solar occultation sensors, and a limited TV capability. Such a payload should probably carry appropriate interplanetary experiments for the transit phase. For the second type of payload, where there is greater interest in physical features of the planet, emphasis would be placed upon the TV experiment. This payload would carry a radio occultation system (which is "free" since it is part of the communication system), and a trapped radiation experiment, a magnetometer, and a microwave radiometer. Payloads larger than this are not feasible since, as shown in Section 5 of Volume 3, much of the overall injected weight is consumed in deboost propellant.

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APPENDIX A

STATEMENT OF WORK

(a) The Contractor shall:

(1) On a level-of-effort basis, provide no less than seven thousand three hundred eighty-five (7,385), nor more than eight thousand one hundred sixty-three (8,163) man-hours of engineering - support personnel services. The above effort shall be directed toward the performance of a conceptual design and feasibility study to develop first-generation spacecraft concepts adaptable for long range, long duration planetary missions in the region extending from Mars to increasing greater distances from the Sun. The study shall include the conceptual design of spacecraft systems to accomplish the following: (1) basic flyby missions of the planets Jupiter, Saturn and Neptune, and (2) examination of the growth potential of the basic concepts through the use of a modular design concept to perform orbiter and planetary capsule entry missions. Particular emphasis is placed upon the spacecraft design tradeoff analysis leading to configuration optimization for a range of injection weights which would have the highest probability of mission accomplishment with the following scientific objectives:

(i) Measurement of the spatial distribution of interplanetary and planetary particles and fields.

(ii) Measurement of the salient features of planetary atmospheres, with particular emphasis upon remote measurements from a flyby spacecraft.

(iii) Observations of the planets, i.e., visual, infra-red, etc.

(b) In performance of this study:

(1) Develop spacecraft system conceptual designs to meet the objectives stated under paragraph (a) (1) by accomplishing the following:

(i) Establish the functional requirements for spacecraft systems to perform the missions.

(ii) Forecast the applicable state-of-the-art for the time period considered.

(iii) Perform design tradeoffs as a basis for the rationale employed for design selections.

(iv) Synthesize the appropriate system concepts.

(v) Identify the problem areas and indicate approaches to their solution.

(vi) Review the system concepts in terms of the Mariner Mars '64 and IQSY Pioneer spacecraft system designs.

(2) Provide a description for each of the systems developed under paragraph (b) (1) which shall include, but not necessarily be limited to, the following:

(i) System block diagrams

(ii) Operational sequences

(iii) Weight and power estimates

(iv) System pointing accuracies and orientation maneuvers.

(v) Spacecraft and science experiment internal-external interface compatibilities, including radioisotope thermoelectric generator radiation and thermal effects.

(vi) Redundancy considerations for increase reliability.

(vii) Evaluation of the design variations required for each of the scientific objectives showing the design complexities involved.

(viii) Spacecraft conceptual configurations and launch vehicle(s) general mechanical compatibility.

(ix) Optimization of the spacecraft systems developed under paragraph (b) (1) with the launch vehicle choices showing tradeoffs involved.

(3) Provide descriptions of the subsystem designs studied and the mechanization approaches to be employed. This shall include, but not necessarily be limited to:

(i) Means by which subsystem designs meet the system, or functional requirements of paragraph (b) (1).

(ii) The design tradeoffs considered and the rationale used for design selection.

(iii) The life time reliability design considerations.

(iv) Identification of the problem areas determined and approaches to their solution.

(4) Investigate spacecraft modular concepts which would provide orbiter and capsule entry capabilities to the basic flyby design for potential growth of the system. This shall include, but is not necessarily limited to:

(i) The mechanization feasibility and spacecraft interface compatibility.

(ii) The operational feasibility and basic spacecraft optimality considerations.

(iii) Modifications required to the basic spacecraft to utilize the modular concepts.

(5) Conduct a reliability analysis for system(s) selected which shall include, but is not necessarily limited to:

(i) Long life time missions to the outer planets.

(ii) System/ subsystem reliability assessments.

(iii) Reliability improvement techniques.

(iv) System failure mode analysis to establish probabilities of mission and partial mission successes.

(6) Perform a cost/effectiveness analysis for the selected spacecraft system(s) and launch vehicle(s) combinations to accomplish the intended mission objectives. The basis of the analysis shall be stated.

(7) Prepare preliminary estimates of schedule and cost for that system(s) developed under paragraph (b) (1). Major variations in cost and schedule shall be noted. The cost estimates shall be in the same format as the costing categories for the Mariner Mars '64 as set forth in Section III of JPL Engineering Planning Document No. 296, entitled "Mariner C Reference Information for Future Mission Studies" dated 15 April 1965 (EPD-296).

(8) Observe the following study constraints:

(i) Mission accomplishments shall be during the 1970-80 time period. For state-of-the-art considerations a reasonable lead time shall be allowed prior to the mission opportunity to insure flight-worthy hardware availability.

(ii) Launch system payload capabilities shall be based upon the following data to be furnished by JPL: (1) Injected payload weight versus kinetic energy (C_3) for 100 N.M. parking orbits for six vehicle combinations. (2) Upper stage envelopes, (3) Mechanical interface data, (4) Final stage stabilization mode and (5) Injection guidance errors. For missions considering use of spin stabilization, additional spin-stable upper stages may be used where applicable.

(iii) Operational compatibility with the Deep Space Network (DSN) as described in the JPL Technical Memorandum 33-83 Revision 1 dated 24 April 1964 entitled "System Capabilities and Development Schedule of the Deep Space Instrumentation Facility, 1964 - 68".

(9) Utilize the following reference information:

(i) JPL Technical Memorandum 33-83, Revision 1.

(ii) EPD-296.